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GEOSTATIONARY COLLOCATION: CASE STUDIES FOR OPTIMAL MANEUVERS

by

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Satellite collocation is not a new topic in the space community. The geostationary belt is considered a natural resource, and as time goes by, the physical spaces for geostationary satellites will run out. The Brazilian Air Force plans to collocate its satellite with another two satellites, and this thesis seeks the most efficient method in terms of fuel optimization and operational aspects. A Systems Tool Kit (STK) software simulation was developed that included various longitude collocation schemes for the Brazilian scenario. The STK simulations showed that a relatively low eccentricity can be achieved with a sun-pointing perigee strategy but that fuel utilization was fairly the same in all of the scenarios. The key takeaway is the operational aspect that relates to how often a maneuver is performed regarding thrusters utilization, minimizing failure risks, and operations workload. To make a solid decision on which strategy to take, other factors must be taken into account and will be commented upon in this work.

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GEOSTATIONARY COLLOCATION: CASE STUDIES FOR OPTIMAL MANEUVERS

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Submitted in partial fulfillment of the
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ABSTRACT

Satellite collocation is not a new topic in the space community. The geostationary belt is considered a natural resource, and as time goes by, the physical spaces for geostationary satellites will run out. The Brazilian Air Force plans to collocate its satellite with another two satellites, and this thesis seeks the most efficient method in terms of fuel optimization and operational aspects. A Systems Tool Kit (STK) software simulation was developed that included various longitude collocation schemes for the Brazilian scenario. The STK simulations showed that a relatively low eccentricity can be achieved with a sun-pointing perigee strategy but that fuel utilization was fairly the same in all of the scenarios. The key takeaway is the operational aspect that relates to how often a maneuver is performed regarding thrusters utilization, minimizing failure risks, and operations workload. To make a solid decision on which strategy to take, other factors must be taken into account and will be commented upon in this work.

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LIST OF ACRONYMS AND ABBREVIATIONS

a	orbit semi-major axis
a_c	geostationary real semi-major axis
e	eccentricity
E	east
ECC	orbit eccentricity
GEO	geosynchronous earth orbit
GOES	Geostationary Operational Environmental Satellites
I_{SP}	specific impulse
ITU	International Telecommunications Union
MCS	mission control sequence
N	Newton
NOAA	National Oceanic and Atmospheric Administration
RAAN	right ascension of the ascending node
RE	radius of the eccentricity control circle
RF	radio frequency
STK	Systems Tool Kit software (Analytical Graphics, Inc.)
W	west
WARC	World Administrative Radio Conference
ΔV	velocity difference (final velocity minus the initial velocity)

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I would like to express my gratitude to the following:

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The Brazilian Air Force, to which I devoted 13 years of my life as a fighter pilot.

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I. INTRODUCTION

A. BACKGROUND

From the first satellite insertion into Geosynchronous Earth Orbit (GEO) in 1963 to the present day, the satellite population at the geostationary belt has been increasing and so have the collision hazards. Far from negligible, collision risks are driving the need to plan and execute strategies that allow us to operate many satellites in the same longitude slot, with minimal, or even zero, collision possibilities [1].

In mid-2016, the first military Brazilian satellite, called SDGC, will be launched and inserted on 75º-west longitude. Since there will already be two satellites at this location, a NOAA GOES satellite and a Brazilian operated communication satellite, StarOne, an operational agreement is hypothesized for a collocation strategy between the satellites. This is discussed in Chapter III.A, which recommends a collocation scheme of complete longitude separation.

B. OBJECTIVES

The objective of this thesis is to analyze the proposed collocation longitude station-keeping and suggest alternatives in order to optimize both fuel utilization and command center operations workload.

C. THE RESEARCH QUESTION

What would be the best longitude separation station-keeping mode at the GEO 75º West longitude for a formation of three satellites to optimize fuel and operations workload?

D. SCOPE

This work examines the complete longitude separation collocation strategy. Other strategies could have been analyzed and proposed; however, the original strategy is well suited for collocating three satellites.

E.   LITERATURE REVIEW AND METHODOLOGY

Collocation strategies receive little coverage in the literature. However, the resources available are comprehensive. For example, *The Handbook of Geostationary Orbits* by E. M. Soop is considered the bible for GEO matters. Hengnian Li recently addressed the specific topic of collocation on his book, *Geostationary Satellites Collocation.* We also referenced *Applied Orbit Perturbation and Maintenance* by C. C. Chao, an important book to better understand the perturbations phenomena among the satellites.

Hardacre [2] and Rausch [3], in their Ph.D. dissertations, discussed the collision risks and control strategies for collocated satellites.

Although many papers explain how the various command centers have collocated their satellites around the world, the focus of this work will be on the aforementioned books because every paper and thesis listed them as main references.

The software Systems Tool Kit (STK) will be utilized to simulate a collocation of three satellites in the 75º w slot. The purposes are to study the fuel utilization, and operations workload in a command center.

STK is a powerful tool that can model practically everything on earth, sea, air and space. For space, it can model every phase from the launch of a spacecraft until its reentry. One of its modules, called Astrogator, is a tool for maneuver planning and trajectory design; it can calculate maneuvers and predict orbits [4]. Its creators maintain a website where they release tutorials, discussions and scenarios re-creating real space missions [5]. Therefore, we will use STK's Astrogator to model the satellites and their orbits.

The simulation methodology employed approximated the same conditions for the satellites in all strategies:

- All simulations started from the same point.

In all the simulations, the satellites start from the center of the allocated control slot. The slot, called dead band, sets the limits for the east-west and north-south

excursions of the satellite. The satellites are propagated to the east edge of the slot and then the maneuver cycle commences.

- The edge of the slot is targeted to be the maneuver place.

The motion of the satellites can be seen in Figure 1. The satellite is at a point near the west edge (A) and a tangential thrust is performed, providing a slight eastward acceleration. The satellite is then propagated according to the strategy chosen. At point B it will reverse its movement as the Earth's westward acceleration dominates, and the cycle will end at the west edge of the box (C).
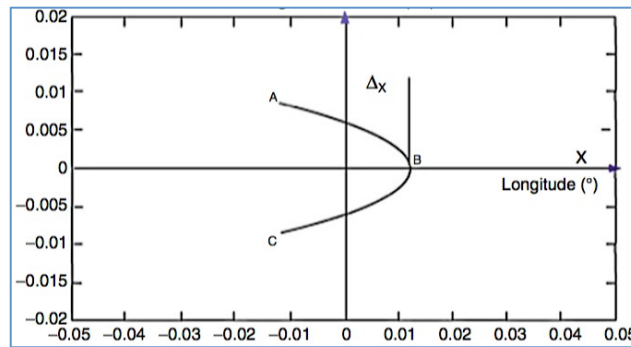
Figure 1.   Satellite maneuver places

The movement of a satellite can be explained by this figure. Starting at point A with a west thrust, reversing the movement at point B, and free drifting to point C. The vertical axis represents the longitude drift rate in degrees per second; the horizontal axis is the longitude relative to the center of the control box. Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 232.

- The same criterion was used in choosing the time to maneuver.

In order to accomplish the sun-pointing strategy to control the eccentricity with just one thrust maneuver, the simulated satellites targeted a time when the sun was in a pre-determined relative position. More on this when we will talk about the simulation eccentricity control in Chapter II.E.

- All the maneuvers within a one-year lifespan were simulated.

The simulations started on January 1, 2015, and terminated on January 1, 2016.

We will calculate the number of maneuvers and fuel utilization by using this methodology. From these values, we can infer our results.

F.   ORGANIZATION OF STUDY

To understand collocation mechanics and strategies, some principles applying to the geostationary orbit must be explained. Although this knowledge is common to every other orbit, we have to highlight the features that make this kind of orbit unique.

In light of this, the basic required information to understand GEO station-keeping will be presented in the theoretical framework (Chapter II).

The simulation analysis in Chapter III will present how STK was used and evaluated.

Finally, the results will be presented and some conclusions will be stated.

II. LITERATURE REVIEW AND THEORETICAL FRAMEWORK

In this section, we will explain some crucial knowledge for understanding the GEO and how its characteristics will influence the way we control geostationary satellites.

First, we will summarize some basic orbit concepts. Second, the GEO will be explained along with its legal framework. Finally, an extensive section on GEO perturbations will be presented, followed by an explanation of the station-keeping maneuvers.

This chapter will create a solid foundation to better understand the simulation that was built in the STK software.

A. KEPLERIAN ORBIT

Keplerian Orbital Elements are the most common parameter set to describe fundamentally conic orbital motion. Rausch [3], provides an explanation of the classic Keplerian orbital elements, (illustrated in Figure 2):

- Semi-major axis - a

The dimension of the orbit; it is half the distance between perigee and apogee. Perigee and apogee are, respectively, the earth orbit's point of closest and farthest approach.

- Eccentricity - **e**

The magnitude of eccentricity vector dictates the shape of the orbit. Its direction is along the line of apses towards perigee.

- Inclination - **i**

It is defined as the angle between the angular momentum vector of the orbit and earth's equatorial plane.

- Right ascension of the ascending node - Ω

It is defined as the angle between the vernal equinox and the ascending node of the orbit. The nodal point is the intersection of the equatorial plane and the orbital plane.

- Argument of perigee - ω

It is the angle measured from the ascending node to the perigee.

- True anomaly - ν

It is the angle between the perigee and the satellite.
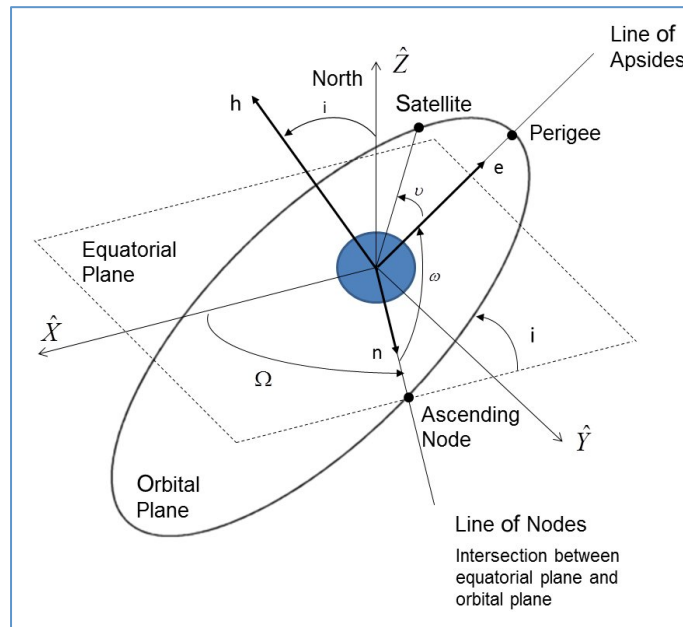
Figure 2.   Keplerian Orbit Elements

The classical orbit elements are depicted in this figure. Source: [3] R. R. Rausch, "Relative orbit control of collocated geostationary spacecraft," Ph. D. dissertation, Purdue Univ., West Lafayette, Indiana, 2012.

1.   Equinoctial Orbit Elements

Keplerian orbit elements are not suited for geostationary orbits because they suffer from two peculiarities. Because the GEO is near circular and near equatorial, it is hard to define the ascending node; hence it is also difficult to define argument of perigee.

Thus, geostationary orbits are most commonly represented in terms of equinoctial orbit elements [3].

Rausch also presented some equinoctial elements. Here is what we will refer in this study [3]:

- Eccentricity vector

$$\overline{e} = \begin{pmatrix} e_x \\ e_y \end{pmatrix} = \begin{pmatrix} e.\cos(\Omega+\omega) \\ e.\sin(\Omega+\omega) \end{pmatrix} \tag{1}$$

- Inclination vector

$$\overline{i} = \begin{pmatrix} i_x \\ i_y \end{pmatrix} = \begin{pmatrix} i.\cos(\Omega) \\ i.\sin(\Omega) \end{pmatrix} \tag{2}$$

The inclination vector points in the direction perpendicular to the ascending node. The eccentricity vector follows the normal convention. Both vectors project into the equatorial plane.

B.   THE GEOSTATIONARY ORBIT

This section will describe the geostationary orbit, explain its characteristics, and how it is legally dealt with nowadays.

The geostationary orbit is a particular orbit around Earth, first envisioned by C. Clark in his 1945 article "Extra-terrestrial Relays." Clark proposed that three satellites, 120º apart, could provide nonstop communications on Earth [7]. This orbit has been used for more than 50 years for terrestrial communications and communications relay between satellites and their associated ground stations, weather observation, and various other purposes.

Li [6] summarizes geostationary orbit characteristics with five qualities:
- The period is equal to one sidereal day

In order to match Earth's angular velocity, the period has to be equal to the time Earth needs to rotate 360º, called a sidereal day. A normal 24-hour day, or 86,400 seconds, is called a solar day, and accounts for the translation movement of Earth around the sun. From noon to noon, Earth rotates 360.985647º. Thus, the sidereal day is less than a solar day by 336 seconds, making the period of the GEO equal to 86,164 seconds.

- It is a circular orbit.

By obeying Newton's law, the centrifugal force acting on the spacecraft must be equal to the attraction force by Earth. From this perspective, we can calculate the theoretical geostationary orbit radius:

$$r = \sqrt[3]{\mu/\varpi_e^2} = 42,164.2 \text{ Km} \tag{3}$$

where:

μ is the gravitational constant equal to 398,600 km^2/s^3.

ω_e is the angular velocity of Earth equal to 7.292115E-5 rad/s.

- The geostationary orbit defines the spacecraft velocity.

From the theoretical radius of the geostationary orbit and Earth's angular velocity, we can derive the geostationary velocity:

$$V_g = \varpi_e r = 3,075 \text{ m/s} \tag{4}$$

- The orbital plane is the equatorial plane.

This is a requirement of the orbit. A true geostationary orbit would have an inclination of zero value, and would exactly match the Earth's equator. However, there is no perfect geostationary orbit, because we have forces that perturb the orbit; the satellite would cross the equator twice each orbit period like an ordinary orbit.

- The nominal longitude is the only complimentary parameter.

Luckily, the only parameter that one can choose is the nominal longitude and it would define a sub-satellite point of interest on Earth.

1.   Legal Framework

The geostationary orbit was declared a natural resource by the ITU International Telecommunications Union on 1971. The ITU was created by the United Nations in 1947 to control RF spectrum utilization. Nowadays, it controls not only frequency allocations but also controls the designation of the nominal longitude of GEO satellites.

Every few years, the ITU promotes a WARC (World Administrative Radio Conference) to define new longitude and frequency leases. Since GEO is a limited natural resource, every country has the right to claim a part of it, even if the country does not have space capabilities.

Initially, the longitude slots were allocated to separate signal frequencies and to avoid interference, regardless of the risk of collision. Thus, the same longitude slot was assigned to many satellites. It was thought that in a shared dead band box 100 km wide in longitude and latitude, the collision risk would be negligible. Thankfully, no collision has been reported yet, but many space agencies have stated that the possibility of a collision is far from negligible and is a major risk to satellites and orbital debris creation.

C.   GEOSTATIONARY ORBITS PERTURBATIONS

This section will discuss the natural perturbations affecting GEO.

All satellites have perturbations in their orbit, but geostationary satellites are more prone to these perturbations. Although very small, if these perturbations are not taken into account, the whole objective to make the satellite "hover" over a specific point would fail.

To better understand the influence of the perturbations on the mean orbit *keplerian* elements, it is common to divide them into two parts [8]:

- Short-term – have period of a sidereal day or shorter. It is not realistic to compensate for them, so they will define the smallest size of the control box.

- Long-term – they change the orbit elements, and are the ones that we take into account for station-keeping.

Both short and long-term forces create motions that influence the movement of the GEO satellites, but it is the long-term ones we should seek to compensate. In summary, Soop [8] stated the primary perturbations and their relative long-term change in the orbital elements:

- The non-spherical part of the Earth's gravitational attraction changes the longitude drift, and has a little contribution to the inclination vector drift.

- The gravitational attraction of the sun, and the moon changes the inclination vector, and the eccentricity vector slightly.

- The solar radiation pressure changes the eccentricity vector.

1.   Earth's Non-Spherical Perturbations

Earth is an oblate, spherical-shaped planet. The oblateness and the elliptic equator plane create extra accelerations towards the radial direction that comes from the zonal term of Earth's gravity, called J_2. Taking the acceleration into account, a real semi-major axis of the orbit is defined. If we were able to maintain our satellite with that semi-major axis, it would be in a stable orbit [9]. Figure 3 illustrates this concept.
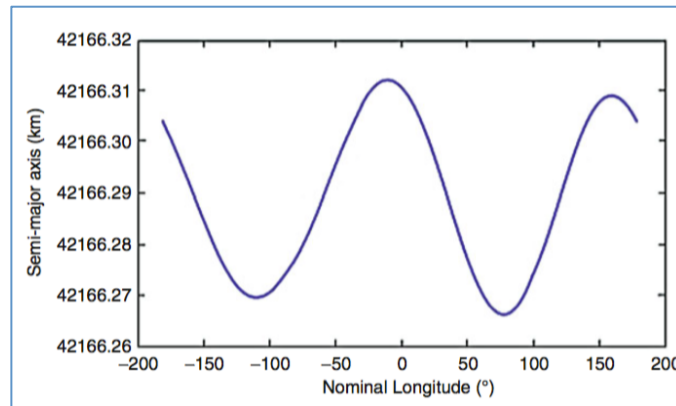
Figure 3.   Real semi-major axis given a longitude

This figure depicts the real semi-major axis with J_2 perturbations at selected longitudes. Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 114.

Unfortunately, simply targeting the real semi-major axis is not enough to keep our satellite stable. Earth along its equator is slightly ellipse-shaped with unsymmetrical mass distributions inside it [10]. This will create zones where a satellite experiences either an increase or a decrease in the gravitational acceleration; this feature is also called triaxility of the Earth. The dominant tesseral term called J_{22} induces an extra gravitational acceleration on the tangential direction of the satellite and creates variations on the semi-major axis that varies in magnitude and direction. For a satellite stationed at 75º W, the triaxility of Earth will increase the semi-major axis by 91.06 meters per day [9].

The variation of the semi-major axis will create a difference between the orbit's angular velocity and Earth's angular velocity; this is called the drift motion of the mean longitude, or just longitude drift. This drift acceleration is specific for each nominal longitude and will dictate to which direction the satellite will naturally drift. Figure 4 depicts the drift acceleration given a nominal longitude.

Figure 4.   Drifting acceleration given a longitude

Source: [6] H. Li, *Geostationary Satellites Collocation*. New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 125.

In Figure 4, there are four points where the acceleration is zero. These nodes are equilibrium points where a mass can stay at rest forever with respect to the longitude. These points can be either stable or unstable. For a stable point a small longitude

deviation from the node, will force a drift back to it. For an unstable point, a deviation will induce a drift away from it. Table 1 outlines their locations:

Table 1.   Equilibrium points

Stable	Unstable
75.1º E	11.5º W
105.3º W	161.9º E

These are the equilibrium points of the geostationary belt. They are also depicted in Figure 5.

The daily rate that the longitude will drift can be explained by this formula extracted from Li [9]:

$$D = -0.0128^{\circ}(a - a_c) \tag{5}$$

where:

a   is the semi-major axis of the satellite

a_c is the real semi-major axis due to perturbations.

Therefore, one km higher than the nominal orbit semi-major axis would create a westward drift rate of 0.0128º per day.

The semi-major axis perturbation grows linearly, and the rate is defined by the J_{22} term. Therefore, in the vicinity of the satellite nominal position, the perturbation of the mean longitude can be considered linear, thus our simulation will use the value for 75º W. Table 2 will summarize the tangential and longitude drift acceleration values:

Table 2.   Tangential and longitude drift acceleration

Nominal Longitude (º)	Tangential Acceleration (m/s^2)	Longitude drift accel. (º/day^2)
-75.00	3.87E-08	-0.00117595

Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, Table 4.5, pp. 128.

Figure 5 will summarize the longitude drift motion of a GEO satellite:
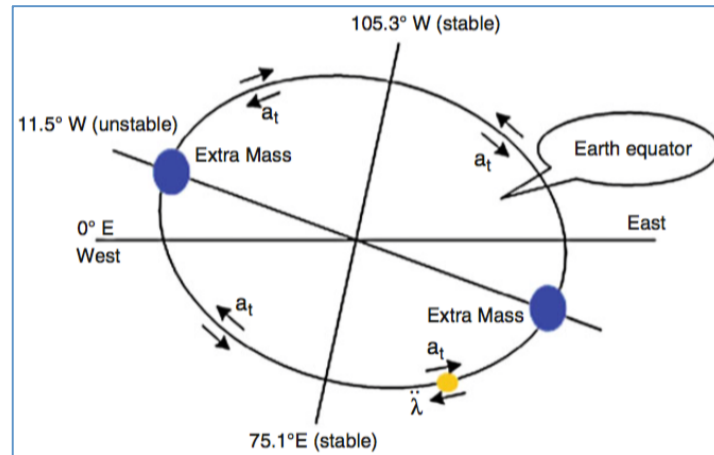
Figure 5.   Longitude Drift motion

This figure displays the movement of a satellite in GEO. If a satellite is located at 0° longitude, it will drift to the nearest stable point, which is th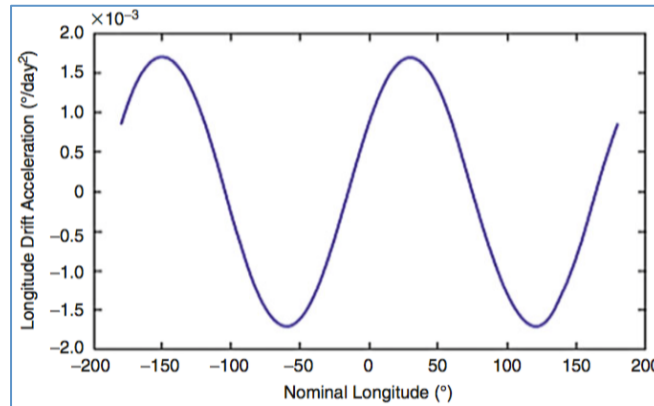e 75.1° E longitude. Source: [6] H. Li, *Geostationary Satellites Collocation*. New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 124.

The non-spherical perturbation also creates a normal acceleration on the orbit and is responsible for its orbit plane precession. This acceleration creates a westward drift of 4.9°/year, with a period of 73.64 years [9].

The eccentricity is also perturbed, but since the satellite is at the same nominal longitude it suffers the same and constant perturbations throughout the whole orbit. There is no long-term perturbation, but a short-periodical daily perturbation acts on the satellite, with amplitude of approximately 3.72E-5 [9].

2.   The Solar and Lunar Perturbations

Gravitational attractions from the sun and the moon also influence the geostationary orbit. Because they are far from Earth they can be considered point masses, however, their forces cannot be discarded. Despite the size of the moon relative to the sun, the attractive force of the moon is twice as big as the sun's. This is due to the fact that the net attraction decreases with the cube of the distance [8]; even though the sun is massive it is distant from Earth.

A GEO satellite would experience the same lunisolar attraction forces as Earth. On Earth, these forces induce the spring and the neap tides; the spring tide occurs during the maximum net attraction. Thus, the satellite would experience maximum forces at the spring tide, and minimum at neap tide.

The lunisolar forces work the opposite way of the Earth's attraction: they decrease the nominal semi-major axis. These forces cancel out during the day, so they do not build up as a long-term perturbation. They do create short-term variations fluctuating from three kilometers during spring tide to one kilometer during neap tide [9].

The satellite's eccentricity vector is also influenced in a medium-term perturbation that varies in a circular motion. This circular motion has the same period as the moon's orbit around Earth, 27 days, and takes an eccentricity radius of 3.5E-5. This medium-term circular motion is identified by the "loop" motion of the eccentricity vector (Figure 6). The sun has a similar effect, but it is masked by the solar pressure perturbation, which will be discussed in the next section [8].
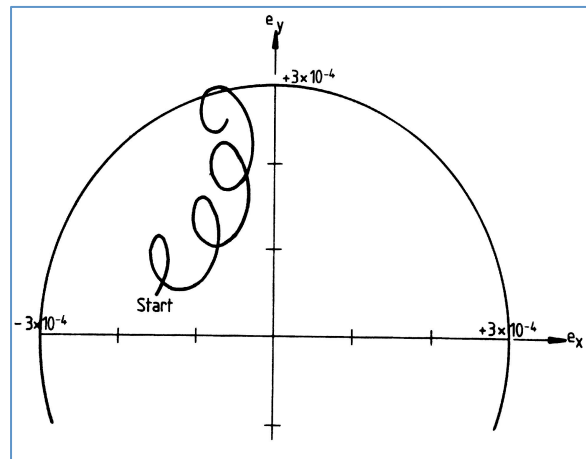
Figure 6.   Eccentricity vector motion

The circular motion of the eccentricity vector is depicted. The four small loops are caused by the moon attraction perturbation. Source: [11] E. M. Soop, *Handbook of Geostationary* Orbits, 3rd ed. Dordrecht, Netherlands: Kluwer Academic Publishers; Torrance, CA: Microcosm Inc., 1994, pp. 84.

The perturbations on the inclination vector are the major effect of the lunisolar attraction. Understanding the effect is quite complicated, but for every practical purpose, the perturbations create a wavy motion on the x component of the inclination vector and will always drift perpendicular to the vernal equinox direction on the y component [9] . In Figure 7, we can see the wavy motion and the growth on y-axis; the inclination started at zero and is propagated throughout one year.
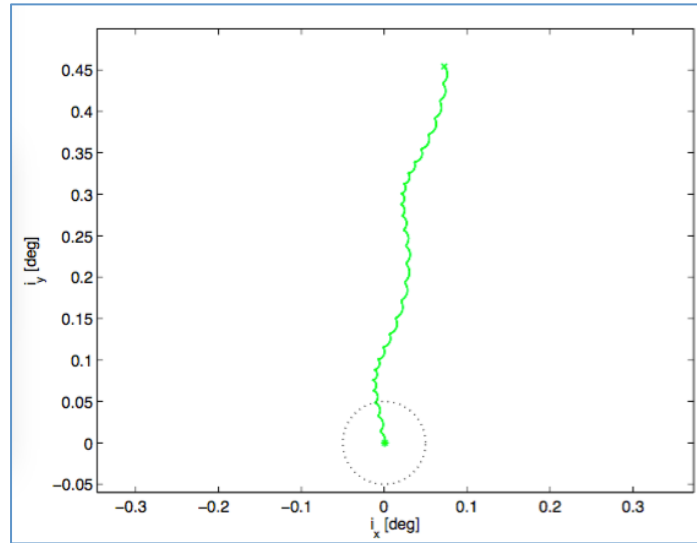
Figure 7.   Inclination evolution

The figure shows an inclination vector, i.e., the orbit's angular momentum vector, drift throughout one year. They are depicted in equinoctial elements. Source: [3] R. R. Rausch, "Relative orbit control of collocated geostationary spacecraft," Ph. D. Dissertation, Purdue Univ., West Lafayette, Indiana, 2012.

The inclination vector magnitude will always increase and where it starts does not make much difference regarding the fuel optimization. We can see in Figure 8 that for an inclination starting at 0.2º with three collections of RAAN (250º, 270º, and 290º were chosen for clearer information), the motion follows the same pattern in all cases. Each satellite was propagated for the first half of 2015.
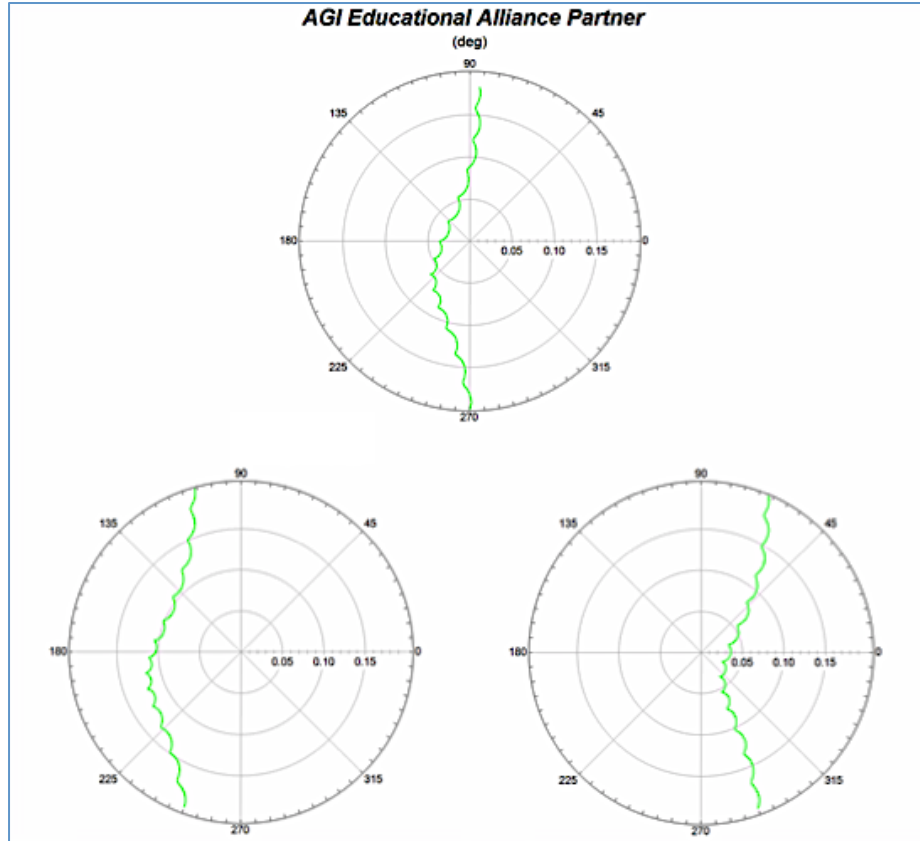
Figure 8.   Inclination drift

These graphics depicts a polar graph of the inclination vector and the RAAN. The center represents a zero inclination value. The radials represent the inclination vector magnitude. A satellite with 0.2° of inclination is depicted, and from left to right, we have respectively 250º, 270º, and 290º of RAAN. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

One consequence of this motion is that the amount of velocity to control the inclination is about the same; regardless, the number of maneuvers made or where the cycle starts. However, slight differences in fuel usage may be found depending on the size of the control box and the level of complexity of the fuel optimization software [13].

3.   The Solar Radiation Perturbation

The solar radiation pressure deforms the shape of the orbit of our satellite. It is created by the electromagnetic radiation of the sun, and it is not to be confused with solar

16

wind. Figure 9 shows that the acceleration due to solar radiation pressure is decomposed into radial and tangential components and exemplifies its effects on the shape of the geostationary orbit [3].
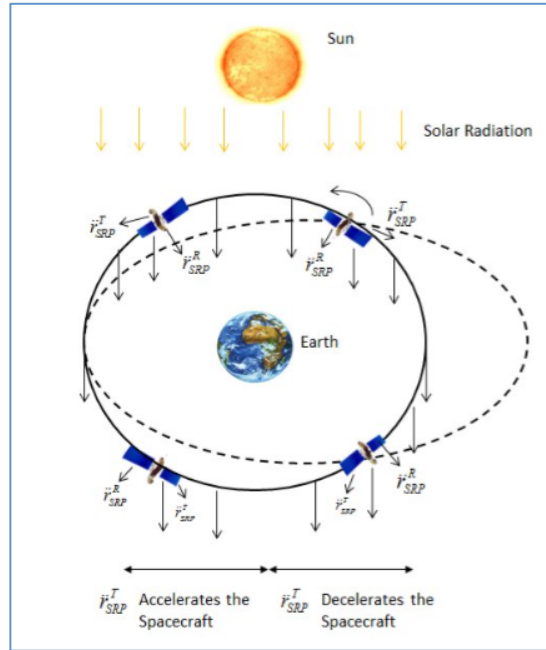
Figure 9.   Solar radiation pressure effect

The figure shows an exaggerated representation of the solar radiation pressure (SRP) acting on a satellite throughout its orbit. Source: [3] R. R. Rausch, "Relative orbit control of collocated geostationary spacecraft," Ph. D. dissertation, Purdue Univ., West Lafayette, Indiana, 2012.

Although it is not the solar wind, the solar radiation perturbation is highly related to the cross-section area of a satellite; therefore, you can relate how the solar pressure works to a wind sail. The force will be greater as you increase the area of the sail. The solar pressure is not a great force; the maximum acceleration value is 3.4E-7 m/s^2 [9]. However, this acceleration is constant throughout the life of the satellite, and therefore cannot be neglected.

Li indicates that the acceleration expression for the solar pressure is [9]:

$$a = -\varepsilon C_R \left(\frac{S}{m} \right) P_o \mu \qquad (6)$$

In the equation:

$\varepsilon = 0, 1$: either the satellite is on eclipse or not;

CR: radiation pressure coefficient relates to the surface material compounds;

Po: solar radiation pressure constant per square meter, 4.56E-6 N/m2;

S: surface area facing the sun;

m: satellite mass;

μ: cosine direction from satellite to the Sun.

The eccentricity vector drift due to solar pressure is an elliptic motion following the mean solar longitude, and often is considered circular to simplify the station-keeping planning. In Figure 10, we can see the natural evolution of the eccentricity drawing a circle. The evolution is caused by all perturbations including the solar radiation pressure.
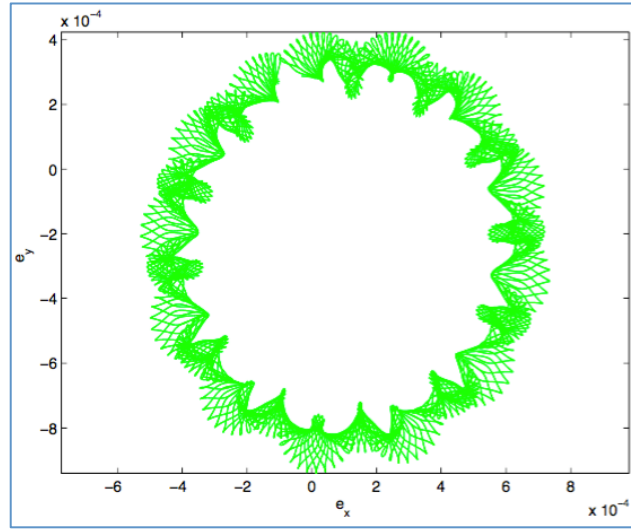
Figure 10.   Eccentricity natural evolution

The eccentricity evolution over the course of one year is depicted on this figure. The eccentricity is represented by equinoctial elements. Source: [3] R. R. Rausch, "Relative orbit control of collocated geostationary spacecraft," Ph. D. dissertation, Purdue Univ., West Lafayette, Indiana, 2012.

Therefore, we often call this the eccentricity perturbation circle, and its radius can be calculated by

$$RE \approx 0.011 C_R \left(\frac{S}{m} \right) \tag{7}$$

The eccentricity perturbation is a superimposed effect and causes long-term perturbations. We will discuss eccentricity further in section E.

The solar radiation pressure even creates a daily short-term perturbation on the semi-major axis and inclination vector, but they average out to zero, and as already discussed, short-term effects are not compensated.

D. SATELLITE STATION-KEEPING MANEUVERS

This section will discuss how a geostationary satellite is maneuvered. The simulation in STK will employ many of those concepts, and it will calculate intrinsically all the equations that will be presented.

We already saw that the geostationary orbit is fairly perturbed, thus if we want our satellite to stay inside a control box we must perform maneuvers to control the longitude drift, inclination, and eccentricity.

This chapter will discuss some basic station-keeping definitions, and the inclination and longitude maneuver. The eccentricity control will be discussed separately because it is a key concept to understand the fuel optimization.

1. Maintaining the Allocated Orbit

The motions of one satellite, or a set of collocated satellites, must be kept within a confined region called a "dead band," or control box (Figure 11), which is defined by the longitude and latitude limits.
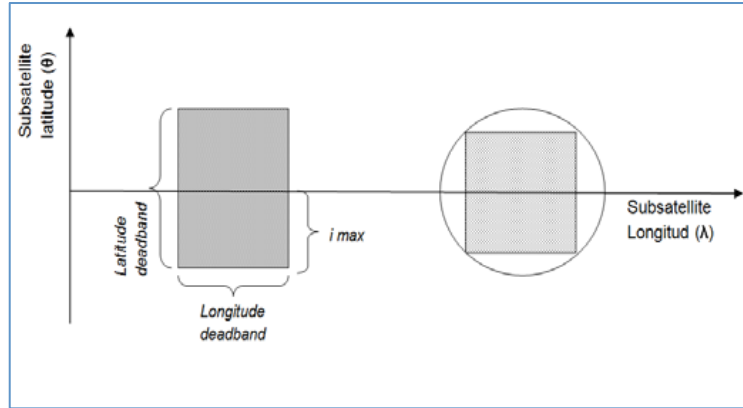
Figure 11.   Dead band

The longitude is shown in the horizontal axis and latitude in the vertical axis. Since the orbit plane is the Earth's equator, one can think of latitude as the satellite inclination as well. Source: [11] E. M. Soop, *Handbook of Geostationary* Orbits, 3rd ed. Dordrecht, Netherlands: Kluwer Academic Publishers; Torrance, CA: Microcosm Inc., 1994, pp. 98.

Therefore, the station-keeping is a collection of planned maneuvers using the satellites' propulsion system to correct the orbit. A small dead band requires more maneuvers and results in more difficult station-keeping. In contrast, a larger dead band results in less frequent maneuvers and simpler station-keeping.

Many approaches are available for planning the maneuvers; we adopt a regular schedule, where every set of maneuvers comprises a station-keeping period. This will simplify planning and optimize the workload of the command center. The following is an example of a four-week cycle, containing two longitude and one inclination maneuver as found in Soop [13]:

Week 1, Monday: Orbit determination and prediction.
                              Check dead band for the next two weeks.
Week 1, Friday: Orbit determination and prediction.
                              Prepare an inclination maneuver for Monday.
Week 2, Monday: Perform inclination maneuver.
Week 2, Tuesday: Quick orbit determination and drift check.
Week 2, Wednesday: Orbit determination and prediction.
                              Prepare a longitude maneuver for the next day.

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Week 2, Thursday: Perform the longitude maneuver.

Week 2, Friday: Quick orbit determination and drift check.

Week 3, Monday: Orbit determination and prediction.

                       Check dead band for the next two weeks.

Week 4, Monday: Orbit determination and prediction.

                       Prepare a longitude maneuver for the next day.

Week 4, Tuesday: Perform the longitude maneuver.

Week 4, Friday: Orbit determination and prediction.

                       Check dead band for the next two weeks.

In our simulation, the schedule will vary according to the strategy adopted.

2.   Inclination Maneuver

The inclination maneuver is the main objective of the north-south station-keeping. In order to keep the satellite inside the allocated north and south latitudes, a thrust in the normal direction of the orbital plane is performed.

Depending on the satellite construction, these thrusters could be installed on a north panel, creating a south thrust; or on a south satellite panel, creating a north thrust. [14, 15]. Regardless of where the thruster is mounted, normal thrusts always change the inclination size and the right ascension of the ascending node. The advantage of having both north and south thrusters installed is that one can plan the inclination maneuver either at the ascending or the descending node of a GEO orbit.

Soop [16] also described very succinct criteria for performing inclination maneuvers

- Maximum inclination

The maneuver is performed to leave the maximum free drift time between maneuvers. It will use the whole latitude box.

- Obeying a schedule, but targeting maximum intervals

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A schedule will be followed, for instance once a month, but the size of the maneuver will be calculated to have a maximum free drift leaving a certain margin to avoid violations on the dead band.

- Obeying a schedule, but the inclination vector size is minimum

The maneuvers will be performed in regular schedule, but the maneuver will target a minimum inclination value between maneuvers.

Whatever option we chose, the fuel utilization will be same, because we have already shown that regardless of when a maneuver is decided upon, the inclination path would be perturbed the same way, so the fuel consumption will be the same with a very good approximation [16].

One can calculate the velocity required from the following [17] :

$$i_f = i_o \pm \left(\frac{\Delta v}{V_g} \right) \tag{8}$$

where:

i_f is the final inclination.

i_o is the initial inclination.

V_g is the geostationary nominal velocity, 3075 m/s.

The sign change in the equation is to account for where the maneuver is performed. A plus sign stands for the ascending node, and the negative sign for the descending node.

From the last equation we can derive a simple summary table of some useful values in inclination maneuvers:

Table 3.   Inclination maneuver summary

To achieve	You'll need
Δi = 1º	ΔV = 53.66 m/s
Δi = 0.0186º	ΔV = 1 m/s

Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 225.

Therefore, this station-keeping is much more expensive than the longitude station-keeping; the ΔV required in a nominal 0.1º latitude dead band averages 47 m/s a year [18].

Nowadays, many GEO satellites at the end of life abandon the inclination station-keeping in order to prolong the operations a little longer by saving fuel. This can be done if the variations on the inclination values are not a problem for the tracking system and other collocated satellites.

3.   Longitude Maneuver

The longitude maneuver is the primary objective of station-keeping maneuvers. In contrast to the inclination maneuver, without longitude station-keeping, any satellite would drift freely to one of the stable points, and the whole objective of offering constant coverage over a nominal longitude would fail.

The longitude perturbations are composed of two elements: the drift due to Earth's non-spherical perturbations, and the eccentricity perturbation due to solar pressure. These elements act independently, but they both cause the satellite drift away from the nominal position [17].

The mean longitude, disregarding the short-term librations and eccentricity librations, will describe a parabola as a function of time. The following sets of equations describe the motion of our satellite:

$$\lambda = \lambda_o + \dot{\lambda}_o t + \frac{1}{2}\ddot{\lambda}t^2 \qquad (9)$$

$$\lambda_o = \lambda_m - \delta\lambda \tag{10}$$

$$\dot{\lambda}_o = 2\sqrt{-\ddot{\lambda}\,\delta\lambda} \tag{11}$$

where:

$\dot{\lambda}_o$:   is the longitude drift rate, D (1), at time zero.

$\ddot{\lambda}$:   is the longitude drift acceleration, mentioned in Table 3.

$\delta\lambda$:   is the dead band half-width measured in degrees.

The longitude drift acceleration will describe the opening of the parabola. A satellite located at 75º W have a negative longitude drift acceleration, which draws a right parabola. In Figure 12, a right parabola and the satellite motion is displayed:
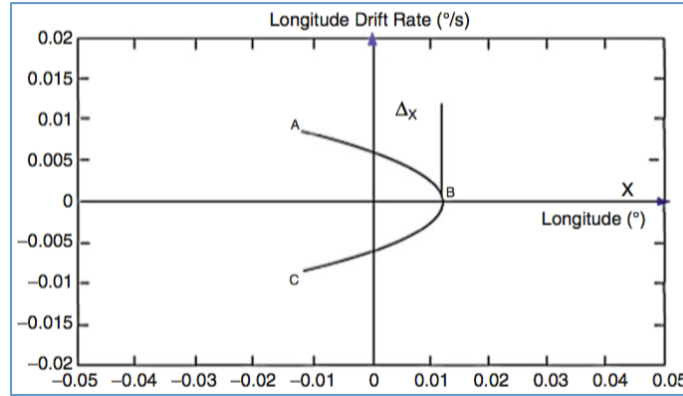
Figure 12.   Longitude drift cycle

Negative longitude drift acceleration creates a motion that draws a right parabola. Starting at point A with a west thrust, reversing the movement at point B, and free drifting to point C. The center of the graph is the center of the dead band. The nominal longitude is on the x-axis and the longitude drift rate is on the y-axis. Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 232.

Every longitude maneuver will also modify the eccentricity vector. Therefore, we have two methods to optimize fuel knowing that both eccentricity and longitude drift are affected. A single thrust longitude maneuver is used for simple fuel optimization. Multi-thrusts maneuvers are used to decrease the eccentricity size, and are often used when the satellite has a great cross sectional area.

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We will discuss eccentricity control strategies in the next section.

In summary, the east/west station-keeping uses tangential thrusts, to change the satellite's longitude drift acceleration to overcome the inherent tangential acceleration perturbation, and to control the eccentricity, so its daily librations would fit within the dead band. This station-keeping uses less fuel than the north/south station-keeping, but is more frequent. It is important to note that the total fuel required is independent of the size of the dead band; it is only dependent on the tangential acceleration at the satellite nominal longitude [17].

In Table 4, a summary of some useful longitude maneuvers operations is shown, and Table 5 shows the calculated velocity increment necessary to maintain our nominal longitude.

Table 4.   Linear relations for east/west station-keeping

ΔV	ΔD	Δa	$|\Delta \bar{e}|$
+1 m/s	-0.352 deg/day	+27.4 km	0.000650
-2.84 m/s	+1 deg/day	-78 km	0.00185

The table shows a typical longitude correction and its changes on the orbit. Adapted from: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, pp. 209

Table 5.   Velocity increment

Longitude	Velocity increment in one year
75º West	-1.22 m/s

Source: [6] H. Li, *Geostationary Satellites Collocation.* New York: Springer; Beijing, China: National Defense Industry Press, 2014, Table 7.3, pp. 252

In Tables 4 and 5, a negative value means a west velocity increment.

As Li explains, longitude station-keeping can be misleading. It does not seem so, but it is more difficult to execute than the north-south station-keeping for the following reasons, according to Li:

- "Tangential maneuvers do not change the satellite's longitude," but they change it indirectly by changing the longitude drift.

- Every tangential maneuver has a coupling effect with the eccentricity, therefore changing the daily longitude librations.

- "The longitude drift rate is highly sensitive to tangential velocity" changes. "The coupling of velocity increment of attitude control and north/south maneuver also changes the longitude drift rate." [17]

Our simulation will not model the last characteristic; we will not change the attitude of our satellite.

Next, we will discuss the eccentricity aspect of the station-keeping.

E.   ECCENTRICITY CONTROL STRATEGY

As noted, the eccentricity will always change with the east/west station-keeping. Because, when a burn is executed, the orbit can be either elevated (if the burn is prograde) or lowered (if the burn is retrograde), establishing a perigee or an apogee, respectively; therefore, station-keeping thrusts turn the orbit into an ellipse. Moreover, the solar pressure perturbation is a constant force that will also deform our circular orbit. The eccentricity itself will normally change regardless of the maneuvers made due to the perturbations already discussed. Figure 13 depicts the natural movement of the eccentricity vector throughout the year. According to equation (7), the size of the control circle will depend on the physical properties of the satellite (shown with a dashed line). This is the circle that the eccentricity will naturally follow due to the solar radiation pressure. However, the gravity perturbations of Earth, sun and moon induce short-term and medium-term variations around the control circle [19].
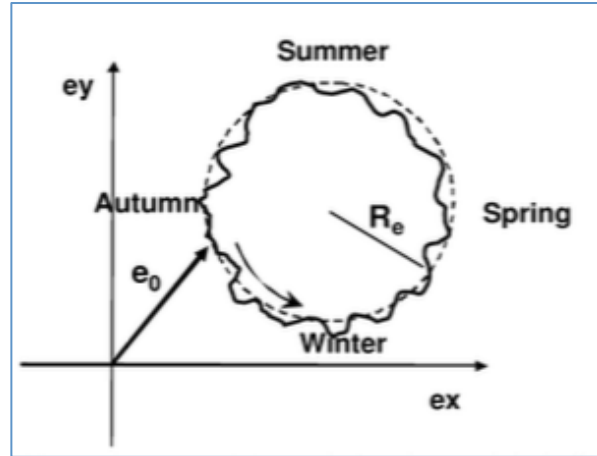
Figure 13. Eccentricity throughout one year

Eccentricity evolution due to perturbations. Source: [19] N. Delong and C. Fremeaux, "Eccentricity management for geostationary satellites during end of life operations," in *ESA/ESOC,* Darmstadt, Germany, 2005, pp. 297–302.

As the eccentricity gets larger it may be impossible to fit the longitude librations within the dead band; Figure 14 exemplifies this problem. The same satellite is propagated from the east edge to the west edge of a dead band. The eccentricity starting values are indicated on each picture. One can see that as the longitude librations increase, the number of propagated days decrease. We started with nine days travel from edge to edge, and finish with two days in the last case. It is worth noting that we could not fit the whole longitude libration in the last case as well.
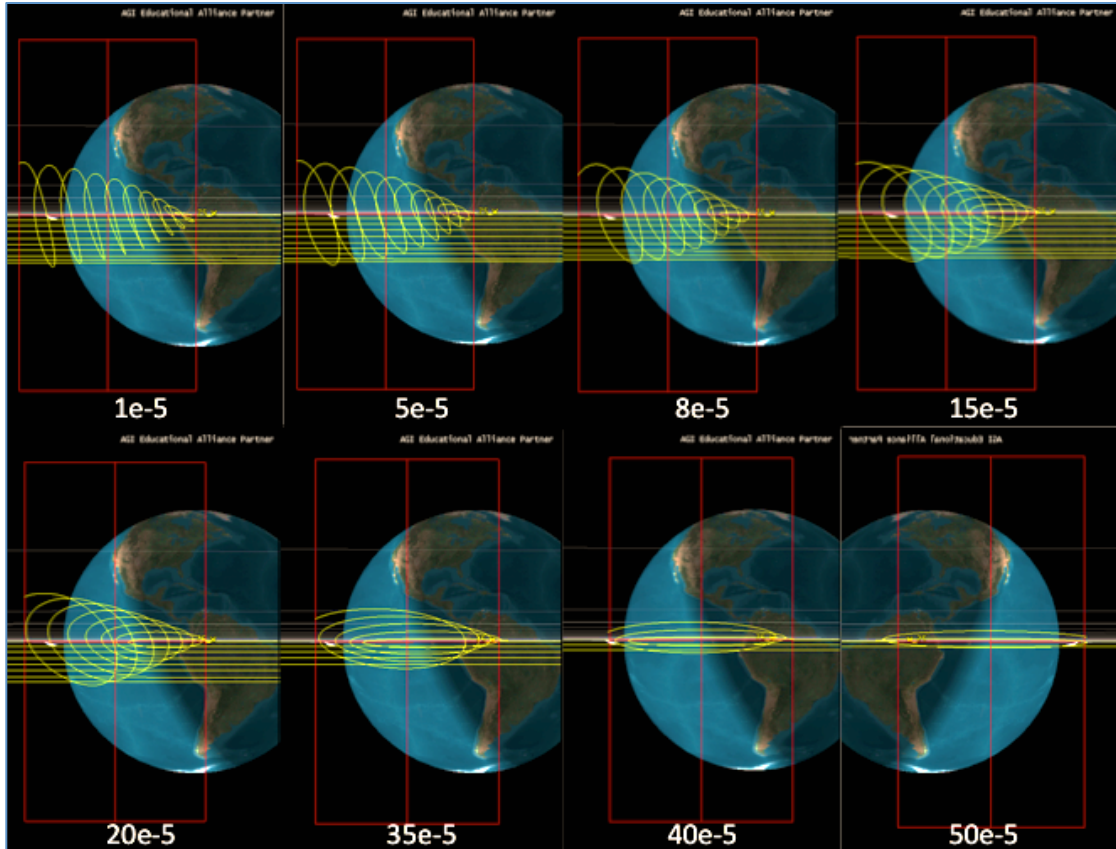
Figure 14.   Longitude librations and ECC

Various values of eccentricity are simulated to show the relation to the longitude libration. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

Hence, we have to control the eccentricity within certain values that allow us to maintain the daily longitude librations inside a control box. We need to choose a place to maneuver that will maintain the eccentricity vector within the control circle. Thus, to optimize fuel, we have to choose the best place to maneuver, and use our east/west thrust to control, firstly, the longitude drift, and, secondly, the eccentricity caused by the solar pressure perturbation.

Eccentricity control can be accomplished with single or multiple thrust maneuvers. Multiple thrusts are used when the satellite has very high solar radiation acceleration where a single thrust cannot account for maintaining the eccentricity vector

inside the control circle. Single thrusts can be used on satellites with low to intermediate cross section to mass ratios [20].

The question of when to maneuver is different for multiple thrusts and single thrust maneuvers. A multiple thrusts maneuver can be done anywhere, because it will use fuel to decrease the eccentricity. Single thrust maneuvers can be performed at perigee, apogee or by establishing a sun-pointing perigee strategy.

In our simulation we use single thrust maneuvers following the sun-pointing perigee strategy, which will be addressed in the next section.

1.   Sun-Pointing Perigee Strategy

This strategy involves an educated study on when to perform a maneuver other than the apogee or the perigee. The idea is to perform a maneuver, and force the eccentricity vector to stay close, either inside or outside, the control circle following the sun position.

Every time a tangential burn is made the orbit's eccentricity is changed. By definition, the eccentricity vector originates at the orbit focus and points normal to the perigee. Using this fact, we will use the solar radiation pressure to control the size of the eccentricity, by making the eccentricity vector, thus the perigee, point to the sun. In theory, the directions must match at the middle of the station-keeping cycle, but in practice, it is only required to point it to the sun sometime during the cycle [20]. Figure 15 shows the eccentricity vector at two particular times during an orbit. The first one is where the maneuver took place. One can clearly see that the eccentricity does not point to the sun. In the second picture, the sun and eccentricity vectors are aligned.
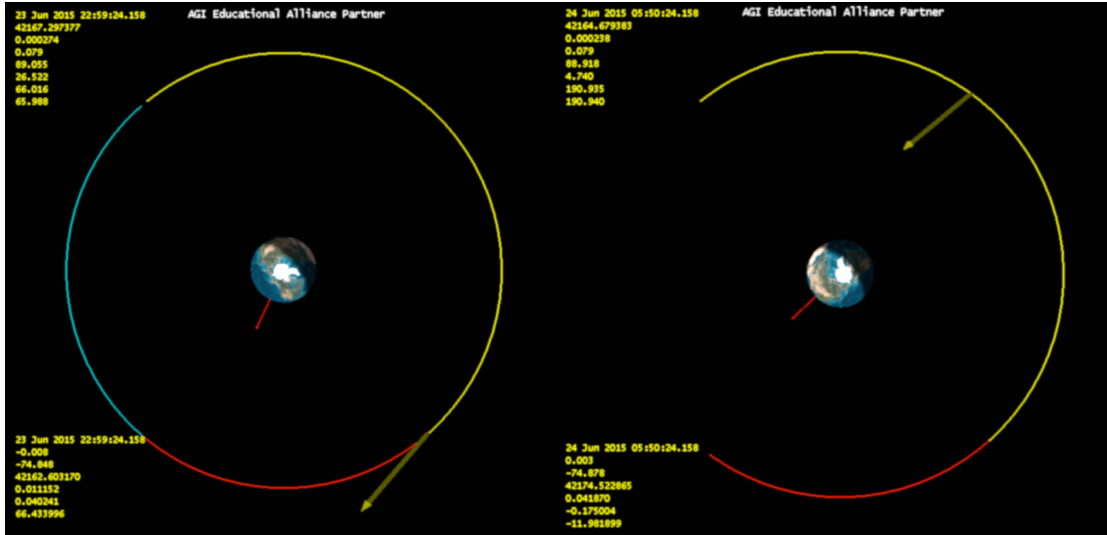
Figure 15.   Eccentricity vector and sun-pointing perigee

The eccentricity vector is shown in red and the sun vector in yellow. The figure on the
left is where the maneuver took place, therefore where the station-keeping cycle begins.
The figure on the right occurred nearly seven hours later and both vectors are aligned,
thus fulfilling the sun-pointing perigee strategy. Source: [12] *Systems Tool Kit*, version
10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

Unfortunately, we cannot make the vector point exactly to the sun throughout the
station-keeping cycle. Therefore, this strategy can only control the size of the eccentricity
by preventing it from far exceeding the control circle. In other words, the eccentricity will
be relatively high, but we can fit the longitude librations within the dead band (Figure 14).

On larger communication satellites, the eccentricity motion due to perturbations is
usually bigger than the control circle; therefore, the eccentricity will increase more than
the single burn at our sun-pointing perigee strategy can control it. Thus, a multi-burn
maneuver must be performed to account for the longitude and eccentricity control. For a
longitude where the longitude drift acceleration is negative, like our satellite, a dual
tangential burn must be made. First, a west thrust must be performed to avoid trespassing
the west boundary, and a half sidereal day later, a small east thrust to decrease the
eccentricity size. The smaller the eccentricity is, the longer is the station-keeping interval,
but the fuel budget will increase [20].

In our simulation we have adopted the single thrust strategy, because of its simplicity, and because we were able to accommodate the longitude librations inside the proposed dead band.

F.   SATELLITE COLLOCATION METHODS

In this section, collocation schemes will be discussed.

The collision hazard on the geostationary belt is well-known in the space community; however, no collision has been reported yet. As the geostationary satellite population increases, the subject of collocated satellites has also received increased attention over the years. With collocation, command centers can now control several satellites within the same dead band, thereby decreasing the probabilities of a collision.

The collision probabilities of two satellites with different inclinations but no separation strategy can be seen in Table 6.

Table 6.   Collision probabilities

Days	Pessimist probability	Optimist probability
1	0.0002	0.00002
4	0.0008	0.0001
16	0.0032	0.0003
64	0.0127	0.0013
256	0.0499	0.0051
1024	0.1852	0.0203
4096 (11.2 years)	0.5592	0.0787

The accumulated risk of collision is shown in this table. Adapted from: [11] E. M. Soop, *Handbook of Geostationary* Orbits, 3rd ed. Dordrecht, Netherlands: Kluwer Academic Publishers; Torrance, CA: Microcosm Inc., 1994, pp. 123.

Soop [13] adds that larger satellites, with large solar panels, will have bigger probabilities of a collision, especially when addressing communication satellites. The

collision would most likely to occur at the tip of the solar array, and because it has a light construction, even a small collision can create serious damage. The motion created by this collision can be controlled, but the differential torque that will result from this damage could possibly cripple the mission in the long run.

There are many possibilities for performing collocation strategies, and the selection of a particular method will depend on distinct characteristics of the satellites and their ground segments. Soop lists a number of features to be considered [13]:

- Number of collocated satellites

- Number of command centers

- Size of the dead band, north-south and east-west

- Duration of the station-keeping cycle, longitude and inclination

- Orbit determination accuracy

- Size of the solar radiation pressure

- Thruster accuracy

- Satellite particular constraints

We have already discussed the scope of collocation. Next, the methods of station-keeping while collocating will be described.

1.   Collocation Approaches

Of all the possible ways to deal with the collocation of satellites, four main approaches are utilized today, and are clarified in Soop [13].

a.   *No Collision Avoidance*

The collision risk is completely ignored and collocated satellites are operated separately. This was commonly done in the past when different control centers controlled collocated satellites. As long as the number of satellites is small, and for short periods of time, this approach may work. That is probably the reason why no collisions have been reported yet.

b. Un-coordinated Collocation with Collision Checking

Each spacecraft is operated to a certain degree of independence, but the centers have to regularly exchange information about the predicted orbits, and before and after maneuvers to check distances between the satellites; a minimum safe distance has to be agreed on. If a collision hazard occurs, one of the operators must modify a scheduled maneuver or perform an avoidance maneuver. After the maneuver is performed, another prediction is made to determine the orbit and check that is safe for other satellites to maneuver.

Since the information flux is intense it is difficult to implement this approach between satellites controlled by different command centers.

c. Collocation by Separation

Each spacecraft must be kept within the assigned dead band. No coordination between centers is required, if they trust that each will maintain their respective dead band. Even though each satellite can be maneuvered independently, a safe margin must be inserted between the dead bands to accommodate errors in the predictions and maneuvers.

Separation can be made by a combination of orbital elements, a subset of the longitude dead band, an area in the eccentricity vector plane, or the latter combined with an area in the inclination vector plane are examples of this method.

The control boxes must be sufficiently large to contain the longitude librations and the longitude drift. This will constrain the number of satellites that can be collocated as the dead band narrows.

d. Coordinated Station Keeping

Coordinated station keeping can be used to optimize the longitude dead band by targeting different offsets like the separation approach: longitude, eccentricity, or inclination and eccentricity combined to different satellites in the same dead band. The main assumption here is that all the satellites would maneuver at nearly the same time,

having the same perturbations integrated over time, thus the inter-satellite distances would be practically the same. A practical way to introduce this method is to set one of the spacecraft as a master with the others maneuvering in relation to it. The flux of information is constant, therefore not suitable for collocated satellites controlled by different command centers.

Independent of the approach chosen, the complexity of operations will grow with the number of satellites and command centers involved. The greatest risk of collision is right after a maneuver, because of inherent thrust errors, and lasts until the next tracking data are collected and the new orbit is determined and predicted. In one of our simulations, this approach will be employed.

2. Collocation Modes

Given the approaches, Soop then details how those are accomplished within a common control box, distinguishing six modes to collocate satellites [13].

a. *Complete Longitude Separation*

The longitude dead band is split into smaller dead bands. It is not a collocation per se, because each satellite operates independently inside its partition.

This method is only possible if the number of satellites is small and if the dead band is wide enough. It is the preferred mode when different control centers deal with collocation.

In our simulation, we will focus on this mode because this is the method proposed on the Collocation Agreement and discussed in the simulation analysis (Chapter III).

b. *Longitude Separation during Drift Cycle*

The longitude separation during drift cycle method also splits the longitude, but now slots partially overlap; different members then occupy the slots at different times of the station-keeping cycle. Longitude maneuvers have to be made on the same day, so they would draw the same longitude drift parabola. This mode is better suited for

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satellites with low solar radiation pressure, because the eccentricity librations are assumed to be small.

We have also addressed one simulation case of this mode to compare some results.

c. Longitude Separation During Eccentricity Libration

The longitude separation during eccentricity libration uses the same set up as the last case, but here the collocated satellites occupy the same region at different times of the sidereal day. All the satellites must be in the same phase of their station-keeping, meaning that they have to maneuver at nearly the same time. The sun-pointing strategy could achieve this requirement, for instance. This method can maintain separation between satellites with high, but almost equal, solar radiation pressure.

d. In-plane Eccentricity Separation

All satellites should use the whole longitude dead band; however, they are separated in radial and longitude directions by a difference in the eccentricity vector. This method could take two approaches: separation and coordinated colocation. The separation approach divides the plane into regions and establishes a safety margin, called the guard band. The eccentricity vector should be targeted within each region. The coordinated approach assigns different target vectors should to each satellite. However, it could use more fuel because more thrust would be required to keep all the vectors inside the maximum eccentricity circle.

This strategy cannot de-conflict the RF spectrum, because twice in one orbit one satellite could possibly block others collocated satellites. Therefore, often an inclination offset is inserted to assure a normal separation for those cases.

e. Combined Inclination and Eccentricity Separation in the Meridian Plane

Since an inclination separation is not enough to separate satellites (they would possibly collide where their orbits cross), we introduce the inclination in one of the

approaches of the eccentricity separations discussed on the last mode. Thus, we would create an out-of-plane separation where the planes cross. The result is that one spacecraft flies around the other in the meridian plane.

This method is suitable for larger clusters of satellites, but it requires the inclination maneuvers to be coordinated and performed at the same time, so it would be easier if all the satellites were controlled by only one center.

In the next chapter, the STK's simulation will be explained in depth.

III. SIMULATION ANALYSIS

In this chapter we will explain how the simulation was created to verify a conjectured collocation strategy, and measure the number of maneuvers and fuel expenditure.

First, we will present the collocation agreement for the 75º-west longitude. Three satellites will be collocated using the complete longitude separation method. For simplicity, some assumptions will be stated and, then, the STK's scenario creation will be discussed. The scenario will depict exactly what the agreement proposes through the theorized station-keeping cycles. Third, the station keeping maneuvers will be modeled. We will explain how they were modeled and some results will be presented.

A. COLLOCATION AGREEMENT

We will imagine a situation where the new military satellite will share the 75º-w longitude with another two satellites. The simplest collocation strategy was picked and the satellites' data will be presented.

(1) Complete Longitude Separation Strategy

Since the satellites belong to independent operators, the strategy based on complete longitude separation is the best choice, provided there are enough margins among the satellites. Because we are dealing with two Brazilian satellites, it is best to keep them together since the command centers can communicate easily. Figure 16 summarizes the concept of longitude separation.
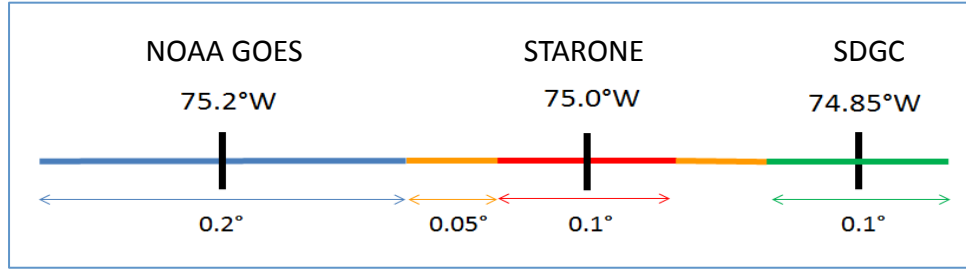
Figure 16. Longitude separation configuration

The satellites and their respectively dead bands are depicted. The gaps in yellow are called guard bands; they are created to improve safety and have the same size, 0.05°.

(2)   Simulation data

The most relevant parameters used for such simulations are listed in Table 7. These inputs are conjectured based on open sources found on the Internet.

Table 7.   Satellite data

	GOES	Star One C3	SGDC
Dry Mass EOL (Kg)	1550	1400	2515
$C_p * S$ (m^2/kg)	0.029	0.030	0.02765958
Longitude box (°)	±0.1	±0.05	±0.05
Latitude box (°)	±0.1	±0.1	±0.1
Nominal longitude (°W)	75.2	75.0	74.85

Adapted from: *Technical Performance Summary*. (2005, Feb.). NASA. [Online]. Available: http://goes.gsfc.nasa.gov/text/GOES-N_Databook/section19.pdf and *StarOne C3 satellite*. Satbeams. [Online]. Available: https://www.satbeams.com/satellites?norad=38991. Accessed Jan. 29, 2016.

B.   ASSUMPTIONS

For all simulations, we assume the following:

- Simulated satellites have the same physical features.

- Thruster firings are considered to be impulsive and no losses were taken into account.

- The satellite has both north and south thrusters installed. This allows inclination maneuvers at both the ascending and descending nodes.

- There are no coupling effects between an inclination maneuver, namely a normal velocity increment, and a tangential velocity increment.

- The inclination maneuver would be performed regardless of any type of eclipse.

- The longitude separation is sufficient to avoid any type of RF interference among the satellites.

C. SCENARIO BUILD UP

The simulation scenario is based on the assumption that we will follow the proposed agreement. The main concern of this strategy is that the satellites would be controlled by three different control centers. As we saw before, this is not a collocation per se, because we will operate the satellite regardless of others in a subset of the whole dead band. Nevertheless, this strategy leaves the decision to each control center to establish the longitude and inclination maneuvers cycles, and of course, any type of fuel and operations optimization.

The idea is to simulate as many different station-keeping cycles as the allocated dead band can hold, and check if we can find any discrepancy regarding fuel utilization. No analysis will be done to study the close approach between the satellites, because we are assuming that each satellite will keep its control box. We will focus on the Brazilian military satellite and its dead band. The nominal longitude assigned to it is 74.85º W and it has a 0.1º interval for longitudinal maneuvers and 0.2º for latitude maneuvers. At the east edge of our dead band, a guard band of 0.05º is included to assure both longitude and RF separation. Figure 18 depicts the scenario and its features.
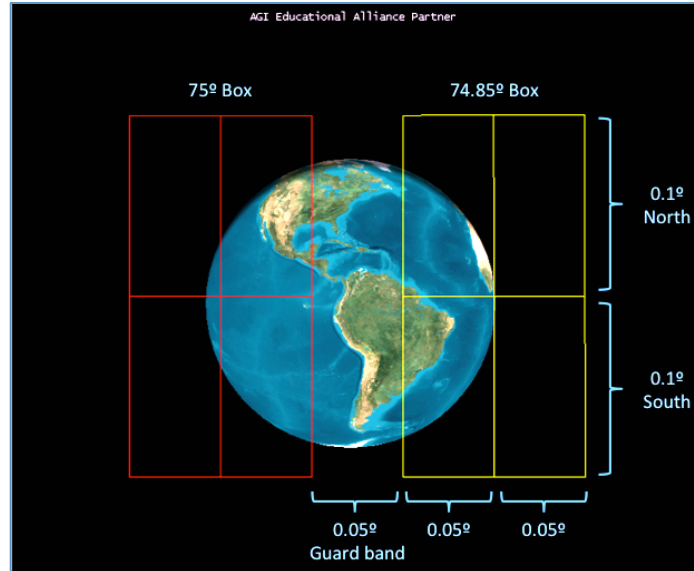
Figure 17.   Dead band of the simulation

This is a STK 3D visualization of the scenario. Source: [12] *Systems Tool Kit, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.*

Within this dead band we simulated four different station-keeping cycles, and all of them use the sun-pointing perigee strategy to control the eccentricity; therefore, they follow the same fuel optimization plan. The cycles follow a regular schedule:

- Seven days longitude station-keeping cycle and fourteen days of inclination cycle (7x14-day)

- Seven days longitude and 21 days of inclination (7x21-day)

- Fourteen days longitude and 42 inclination (14x42-day)

- Twenty-one days longitude and 63 inclination (21x63-day)

The latter station-keeping cycle was simulated using the entire dead band of two satellites, including their guard band; therefore, a cooperation plan must have been agreed for its completion. In later sections, we will explain the 21x63-day cycle separately.

All the station-keeping cycles were simulated during a one-year time span. Thus, only the maneuvers within this time frame were accounted for. In order to have a precise

measure of the ΔV, some station-keeping cycles near the end of the year were propagated until the end of the cycle, therefore extending the simulation length a little.

1.   STK Astrogator

We are going to use a module of the STK software called Astrogator. Its definition can be better explain by the software owner, AGI [21]:

> STK Astrogator is a specialized analysis module for interactive orbit maneuver and spacecraft trajectory design. It supports an unlimited series of events for modeling and targeting a spacecraft's trajectory, including impulsive and finite burns and high-fidelity orbit propagation. Event triggers allow you to focus on overall mission needs without having to perform off-line calculations to set up the problem. Astrogator offers tremendous flexibility through the use of customized force models, engine models and spacecraft attitude, along with the ability to solve for and optimize solutions.
>
> Astrogator provides analysis of the entire trajectory, which can be used as the foundation for an overall STK system analysis (…). During flight operations, you can refine maneuver plans with flight-generated data— such as engine-calibration parameters and actual initial orbits—to produce thruster firing and timing data for command generation.

In this section we present some generic options and configurations utilized by the Astrogator in the simulation.

(1)   The satellite

All the satellites will have the follow characteristics depicted in Figure 18:

Figure 18.   Spacecraft parameters

STK Astrogator control interface. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

The mass of the satellite was chosen to match the average mass between the real players involved in the collocation strategy. The solar radiation pressure coefficient (C_R) and area were chosen to reflect the satellite's assumed values (Table 7). Moreover, the other parameters are default values and do not play a significant role for GEO satellites.

The Fuel Tank tab depicts a typical monopropellant propulsion system. The choice of the engine was driven by the lack of information regarding the engines of the actual spacecraft; therefore, a generic engine was adopted. The software's default *Constant Thrust and I_{sp}* engine has an I_{sp} of 300 seconds and 500 N of thrust. Since we want to simulate an impulsive maneuver, this engine is well suited, although the thrust is unrealistic for a thruster responsible for a geostationary maneuver [14].

Table 8 will complement the satellite features used in some calculations:

Table 8. Spacecraft calculations parameters

Cross section area/Mass ratio	0.0212 m^2/kg
$C_R\left(S/m\right)$	0.0276 m^2/kg
RE	0.000304

These factors are the results of the spacecraft parameters.

Each of the station-keeping cycles will be simulated with this satellite.

(2)   Coordinate System

We will use TOD (True of Date) as our coordinate system, because it is a quasi-inertial reference system utilized for the geostationary orbits, and it contains both the precession and nutation of Earth [22]. The time reference system used is the Gregorian coordinated universal time, UTCG.

(3)   Keplerian Orbit Values

The initial state is the same for all the station-keeping cycles. We tried to release the satellite at the center of the dead band with orbit elements that would be categorized as a neutral position. Therefore, an approximate value of the real semi-major axis was used (Figure 4), and rounded to 42166.5E7 km to expedite the beginning of the simulation; the eccentricity and the inclination vectors were set to zero.

In the following sections, longitude, eccentricity, and inclination controls will be discussed; in each one, a broad approach will introduce the subject and some considerations will be made for each station-keeping case.

D.   LONGITUDE SEPARATION COLLOCATION STRATEGY

To access this strategy, each satellite will maneuver inside an allocated control box, and must remain inside during all its service life. Thus, all simulated satellites will

start from day one at the center of the dead band and be propagated to the east edge, normally taking six days to reach the edge.

The Astrogator target tool is used to compute a west velocity increment that gives us the desired days of the longitude station-keeping cycle. The satellite is then propagated forward in time and the cycle will repeat regularly.

1.   Eccentricity Control

A sun-pointing perigee strategy will be used as a fuel optimization strategy and will control the eccentricity within the parameters (Table 8).

To accomplish this strategy we will follow Soop's steps [20]. For negative longitude drift acceleration, we will maneuver our satellite every time we encounter this:

$$S_b = S_s + \pi/2 \qquad (12)$$

where:

S_b: is the sidereal angle of the satellite, meaning the longitude of the satellite;

S_s: is the sidereal angle of the Sun.

To find this position in STK, we used a Calculation Object called *Local Apparent Solar Longitude*, which gives us the apparent solar longitude minus satellite longitude. Setting the desired value to 90º, we found the exact time to execute the maneuver, which is around 18 hours solar local time [20]. The longitude maneuver would have to be performed there for the sun point perigee strategy to succeed.

We are about to discuss the station-keeping cycles regarding the longitude maneuvers.

2.   Longitude Station Keeping Sequence

Each maneuver in STK is organized inside sequences. STK's help website has the definition for a sequence [23]:

A sequence is a structural element that can be used to organize segments and define the nature of the results that are passed on to the next segment or sequence in the MCS [Mission Control Sequence].

Each sequence represents one longitude station-keeping cycle. For a clearer understanding, the major sequences are subdivided into smaller subsequences.

From Figure 19, we can see that we have another cycle inside, an *EW NS SK* (east/west and north/south station-keeping) sequence; this one stands for a longitude station-keeping and a nested inclination station-keeping, which will be explained in the next section.
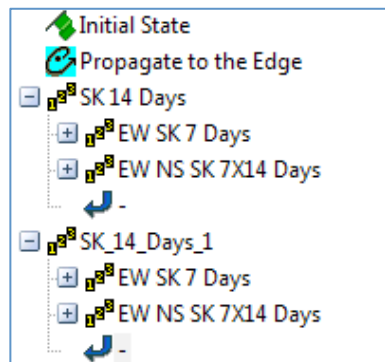

Figure 19.   Station-keeping sequences and subsequences

Subsequences have the commands to execute the mission; inside each longitude sequence one can find:

1.   Propagate to Sun point

The propagate segment is used to find the position where we will have the sun at 90º position relative to our satellite, thus fulfilling the sun-pointing strategy.

2.   Turn around

A target sequence tool is used to find the right amount of velocity to overcome the longitude drift and draw the longitude drift parabola. In STK, the Astrogator target tool

uses a differential corrector to find values for variables defined by the user [23]. Inside the target sequence, we inserted a drift maneuver and a propagate segment. The target tool uses the drift maneuver to employ the thruster, varying the amount of velocity to reach the number of days defined by the propagate segment. The interval of days between longitude cycles is the goal to achieve, but we have to set a tolerance value to simplify and expedite calculations. The differential corrector will use the drift maneuver and the propagate segment as *control parameters*; it will vary these parameters to meet the equality constrains.

In STK the user can seek for certain results at the end of a segment. These results are set as equality constrains, and are used by the corrector as end goals. There are two equality constraints in the propagate segment:

- Longitude – constrain the end of the propagation to a longitude at the east edge of the dead band.

- Longitude drift rate – constrain the target tool to find longitude values at the end of the maneuver congruent to the initial longitude value.

Each longitude constrain is unique in every station-keeping. I could not find a way to automate the choice of the longitude target. Therefore, every east/west station-keeping cycle had to be analyzed in order to come up with a longitude value near the edge of the dead band.

The longitude motion is a right parabola, therefore we will use the longitude drift rate constraint to target the initial longitude. In this way the target tool will have an input that we want to stop the maneuver very close to where we have started it. Thus, we will set the longitude drift rate to 0º/day, set a tolerance, and let STK do the work. Astrogator will use this value as a reference to expedite calculations.

The final longitude station-keeping sequence is depicted in Figure 20.
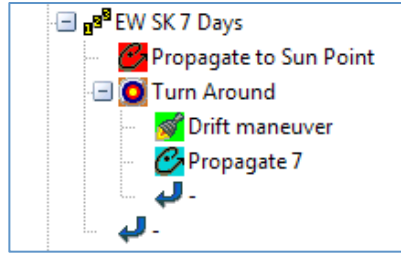
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Figure 20. East/west station-keeping sequence

The color code in the segments will also be used on the orbits tracks in the 2D, and 3D views of STK, thus making it easier to understand and point out every step in the simulation. Here, a red segment means we are propagating the satellite to the sun-pointing place and a blue segment means a longitude station-keeping propagation. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

a.   *Station Keeping Cycles*

The station-keeping cycles lengths are 14, 21, 42 and 63 days. Inside the major sequences, the expectation is for one or two simple longitude station-keeping and one combined longitude and inclination station-keeping. Table 9 will summarize how the cycles are composed.

Table 9.   Station-keeping cycles

Days of SK cycle	14	21	42	63
Number of EW SK and days	1 / 7	2 / 7	2 / 14	2 / 21
Number of EW NS SK and days	1 / 7	1 / 7	1 / 14	1 / 21

"EW SK" stands for a simple longitude station-keeping. Moreover, "EW NS SK" stands for the nested inclination maneuver inside a longitude station-keeping. For instance, the 21 day SK cycle has two "EW SKs" of seven days and one combined "EW NS SK" of seven days.

Every *propagate* segment at the end of a longitude station-keeping target is set to the number of days indicated by its label minus 0.2 days. Thus, *Propagate 7* will have 6.8 days, *Propagate 14* will have 13.8 days, and so on. This action leaves space for the next station-keeping cycle to find the sun-pointing point before another longitude maneuver commences. The east/west and north/south station-keeping will divide the longitude

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propagation into two segments. The sum of the propagated days is always missing one day, which is accounted for in the *propagate to ignition* segment, which is always set to one day.

(1)   63-day station-keeping cycle

The last station-keeping cycle has longitude librations larger than the allocated dead band. Therefore, a cooperation agreement between the command centers would be required. These are the main concerns:

- They would have to exchange orbit determination information regularly and also perform the maneuvers at the same time.

- Their longitude station-keeping cycles would have to be the same, but the inclination station-keeping is optional.

The dead band of the cooperating satellites can be seen in Figure 21; we have deleted the guard band. Although this whole area can be separated into two subareas, the boundary does not exist, therefore the satellites can move through the entire dead band.
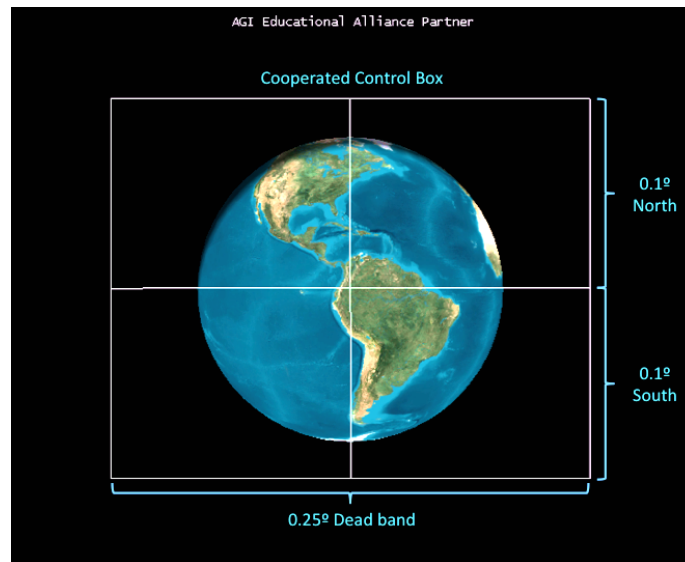

Figure 21.   Cooperation dead band

This figure depicts a 2D visualization of the cooperated dead band. There are no guard bands. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

E.   INCLINATION STATION KEEPING

Regarding fuel utilization, the inclination control is inherently the most expensive feature of a station-keeping. As already discussed, where to maneuver, and which strategy to choose from does not impact the fuel expenditure. However, we will simulate two different maneuvers in each station-keeping cycle:

- Every inclination maneuver will target a zero magnitude of the inclination vector.

- The size of the inclination vector and RAAN will be modified in order to have the least inclination value, and the maximum interval between inclination cycles;

Next, we will explain how the inclination station-keeping will be performed.

1.   Inclination Station Keeping Sequence

The inclination maneuver will be inserted inside a longitude station-keeping, thus the propagation after the drift maneuver will target the beginning of the north-south station-keeping cycle. According to Soop, the inclination maneuver must take place closer to a longitude burn in order to account for thrust errors [16]. Although the simulation does not account for these errors, we tried to abide with this concept of operations and planned our north-south station-keeping to occur three days before an east-west maneuver.

Each inclination sequence starts with a longitude sequence. The inclination target tool sequence will work nested with the longitude target sequence. Thus, STK will iteratively work on both targets to find a solution to fill all variables. The sequence is constructed as follows:

1.   Propagate to sun-pointing

This segment has the same objectives as the longitude station-keeping sequence.

2.   Turn around target

This segment has the same objectives as the longitude station-keeping sequence, except that we have to divide the *propagate* segment into two. The first segment will lead

to the inclination maneuver, and its length depends on the station-keeping cycle in question; the second will be used to propagate the satellite towards the edge of the dead band, and possesses the same equality constrains discussed in the longitude station-keeping sequence.

3.   Nested inclination target

The nested inclination target is consisted of a *propagate to ignition* segment and a thrust *maneuver*. The propagate segment is set to one day, and enables the target tool to vary the time to start the maneuver to find the common point between the old and the new orbit. Therefore, STK can perform noncoplanar maneuvers that change the inclination vector and RAAN altogether [24]. The common points can be either the ascending node or the descending node; the user can choose either one by selecting a positive or negative normal velocity, respectively, in the *maneuver* dialog depicted in Figure 22.
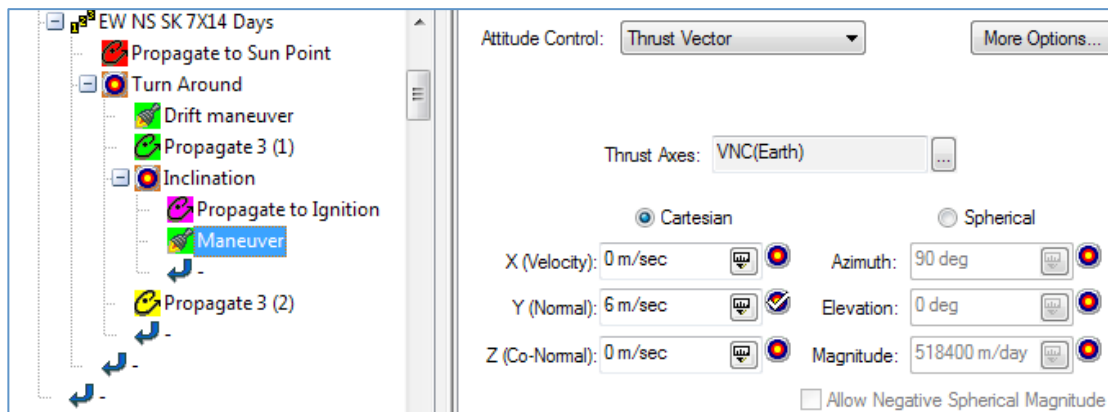

Figure 22.   Maneuver dialog and inclination station-keeping sequence

The color-coding is similar to the longitude station-keeping. Every inclination station-keeping will start with a green segment of the longitude station-keeping. The purple segment will follow indicating the propagation to the ignition of the thrusters. The sequence ends with a yellow segment indicating the propagation to the edge of the control box. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

The inclination maneuver has four different equality constraints that will dictate the type of strategy chosen:

- Inclination TOD

    The size of the inclination vector can be chosen by this constraint.

- Inclination difference

    An inclination difference between the initial and final inclination value can be set.

- RAAN TOD

    A RAAN value can be set; STK will try to accommodate it.

- RAAN difference

    A RAAN angle difference can be set between an initial and final value.

The inclination target tool segment will set the right ΔV regarding the type of inclination strategy defined by these equalities constraints. As noted before, they will dictate a set of inclination maneuvers strategies utilized in the simulation:

1.   Zero Inclination

Inclination TOD is set to 0.001 and RAAN will be set according to size and direction of the perturbations. Therefore, after a brief inclination graph analysis either *RAAN TOD* or *RAAN difference* could be used.

Zero inclination is impossible to get from STK, the differential corrector does not converge, and thus we have to set it to such a small value.

2.   Maximum inclination interval

The maximum inclination interval is used for station-keeping cycles that have greater range of inclination values (i.e., that use the whole latitude band). The inclination vector and RAAN are set to values that give the maximum interval between maneuvers using the *optimal node* concept. This concept can be achieved when the inclination vector, during its free drift, passes through zero [2].

A combination of equality constraints will be used to achieve the maximum interval, and the selection will vary depending on the position and perturbations on the satellite.

a.   Station Keeping Cycles

For all the station-keeping cycles we will execute both inclination strategies. To achieve the ultimate goal of the maximum interval strategy the inclination vector will be targeted to a size that enables the maximum time between maneuvers. All the cycles could perform it without any amendments. The typical inclination maneuver for the maximum interval strategy is shown in Figure 23.
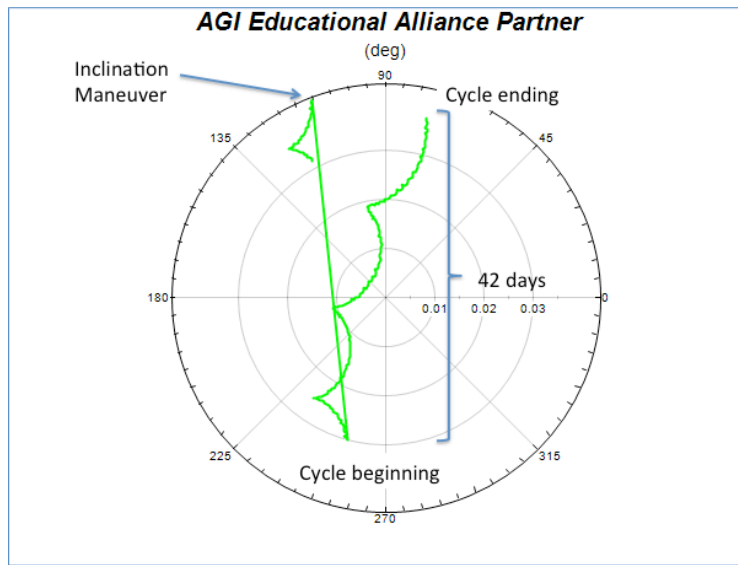

Figure 23.   Maximum inclination strategy

None of the cycle had problems with this method. Here, an excerpt of the 42-day inclination cycle is depicted. Note that the inclination vector at the beginning is targeted to 0.03° with 285° of RAAN; at the end the inclination reached 0.035°. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

On the other hand, the zero inclination strategy had some problems. Cycles of 14 and 21 days are small enough to fit inside the dead band, but on the 42-day cycle, the inclination buildup was too large for the size of dead band. Hence, most of the inclination maneuvers on this cycle targeted zero inclination value, but when the lunar-solar perturbations got their peak, we had to use the maximum inclination method instead.

This action was expected because, as C.C. Chao stated in his book, the inclination interval can be projected by [18]:

$$\Delta T = 98 \sin^{-1}(i/7.5) \tag{13}$$

Therefore, for our zero inclination strategy the "i" value must be substituted by 0.05°, since we only have half of the latitude dead band to work with; the result is a 37-day maximum interval. Because this equation is derived from a perfect geometric relation, one should expect that a 42-day interval would also be feasible.

(1)   63-day station-keeping cycle

The zero inclination strategy had to be abandoned for this cycle. The 63-day excursions of the inclination vector would not fit inside the dead band. Instead, for the sake of testing different approaches to the inclination vector, we have used the second criterion suggested by Soop: we will target the inclination vector so that on the forthcoming cycle it reaches the boundary of the north latitude band [16]. This maneuver strategy is exemplified in Figure 24.
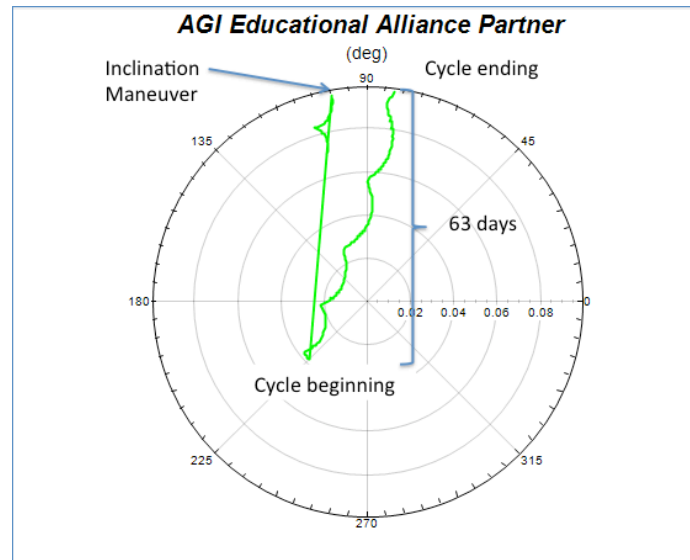

Figure 24.   Maximum inclination method for 63-day cycle

An extract of the 63-day inclination cycle is depicted. Note that the inclination vector at the beginning is targeted to 0.04° with 225° of RAAN; at the end, the inclination approached 0.1°. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

F. RESULTS

This section will discuss the results regarding the station-keeping cycles' maneuvers and their fuel expenditure.

1. Longitude Station-Keeping

The complete longitude separation strategy proved feasible for the proposed scenario. All the satellites remained inside their allocated dead band. As stated, we tried to perform the maneuver to follow a schedule to accommodate the needs of the personnel involved in control operation. Longitude maneuvers would happen every Tuesday and the inclination maneuver on a Saturday or Sunday. This schedule could be adjusted so that the maneuvers could be performed on weekdays; however, it is not within the scope of this study. Figure 25 exemplifies one longitude station-keeping maneuver.
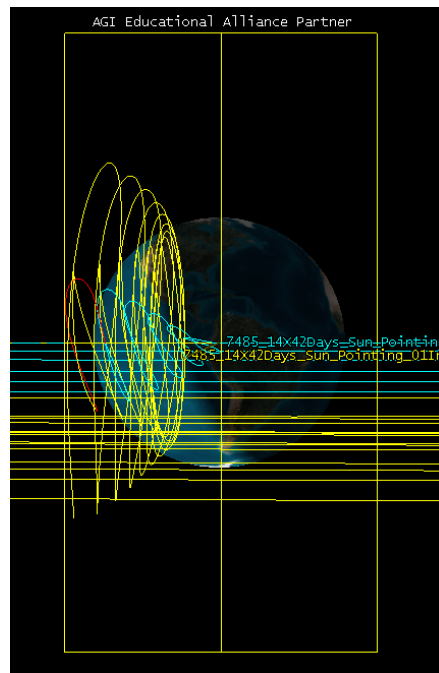

Figure 25.   Longitude station-keeping

The figure displays the first longitude station-keeping of the 14x42-day cycle. In blue and red, one can see the propagation of the satellite to the west border. The yellow path is the Astrogator's propagation of 14 days. It started and ended close to the west border. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

The next table summarizes the maneuvers results for each station-keeping.

Table 10.   Longitude station-keeping summary

Cycle	Maneuvers	ΔV (m/s)	Fuel (Kg)
74.85º - 7x14	52	1.253	0.994
74.85º - 7x21	52	1.259	1.001
74.85º - 14x38	26	1.259	1.001
74.85º - 21x63	18	1.319	1.045
75º - 21x63	18	1.314	1.040
Mean		1.282	1.017
Standard Deviation		0.028	0.021

As expected, the longitude fuel utilization is uniform because the amount of velocity to manage the satellite is related to where it is stationed. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

a.   Eccentricity Control

The indication that the eccentricity strategy worked is the various graphs of the eccentricity vector throughout the whole simulation; they all resemble the natural drift of the eccentricity vector. Since the time of maneuver was the same for each satellite, the graphs are expected to be very similar. The eccentricity was effectively kept within the limits under the sun-pointing perigee strategy and we successfully fit the longitude librations inside the dead band. In the following figures the eccentricity drift of all satellites will be presented. They are presented in equinoctial elements of eccentricity and depict the excursions of the eccentricity vector throughout the simulation time span.
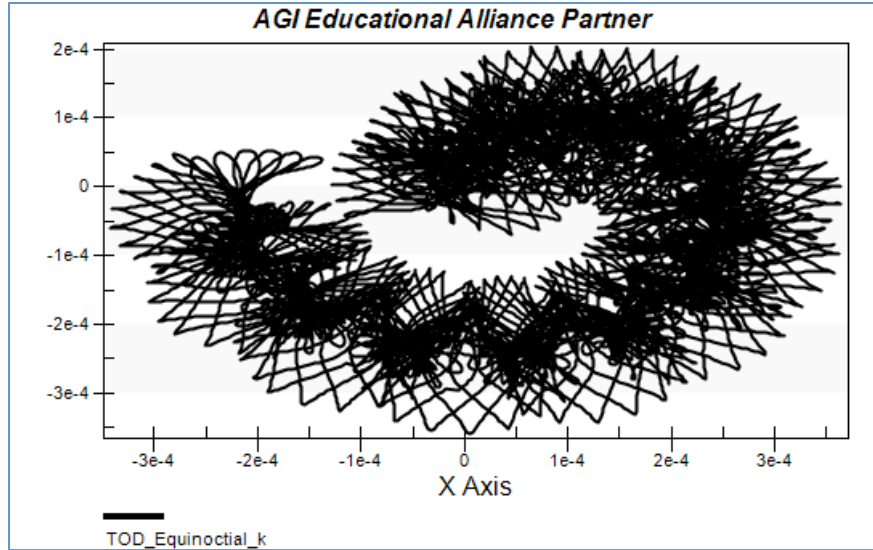
Figure 26.   7x14-day cycle

This graph was created using the equinoctial elements and depicts the movement of the eccentricity vector. Source for all figures: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.
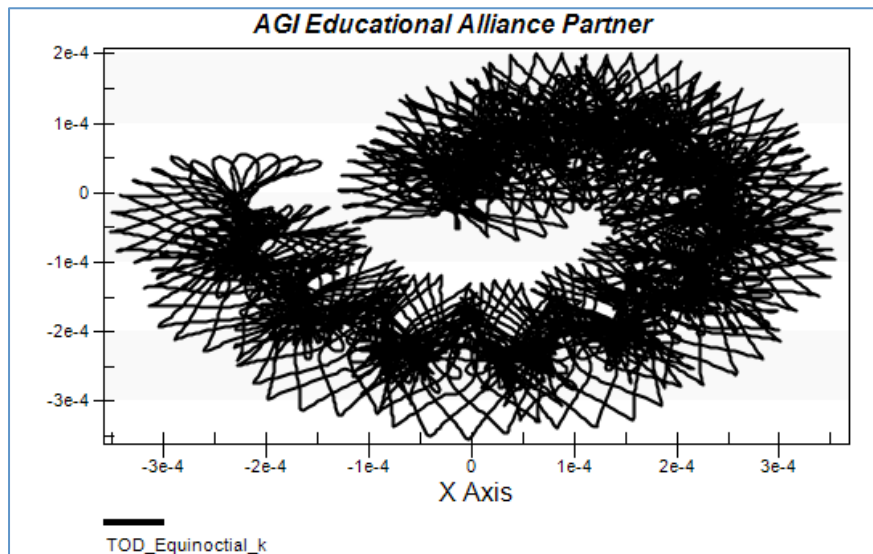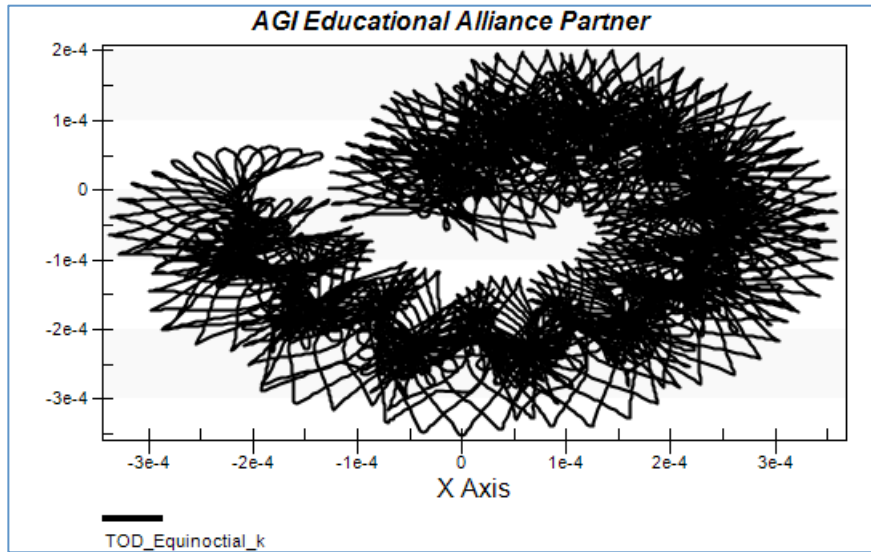

Figure 27.   7x21-day cycle

This graph was created using the equinoctial elements and depict the movement of the eccentricity vector. Source for all figures: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.
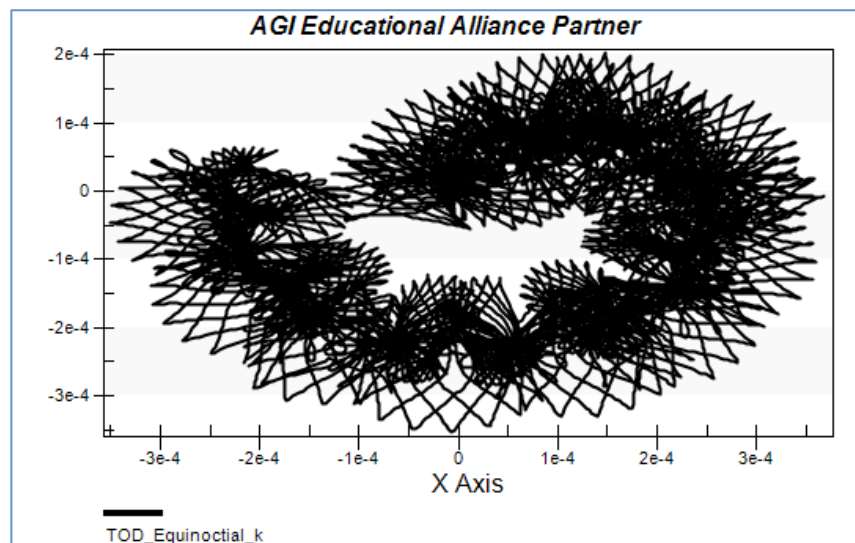
Figure 28.   14x42-day cycle

This graph was created using the equinoctial elements and depict the movement of the eccentricity vector. Source for all figures: [12] Systems Tool Kit, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

This graph was created using the equinoctial elements and depicts the movement of the eccentricity vector. Source for all figures: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

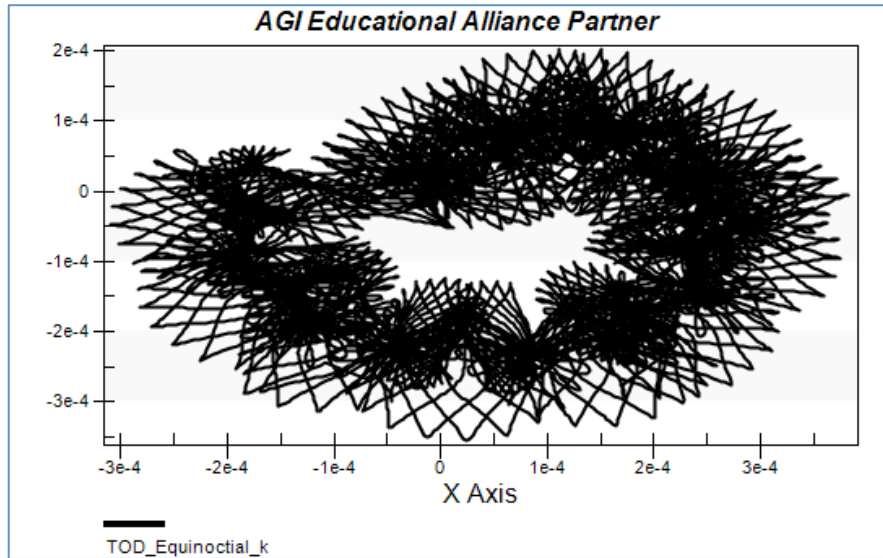Figure 29.   21x63-day cycle – 74.85º satellite

Figure 30.   21x63-day cycle – 75º satellite

The graphs were created using the equinoctial elements and depict the movement of the eccentricity vector. Source for all figures: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

It is worth noting that the above charts do not draw a circle because we started the satellite simulation with zero eccentricity. As stated, the sun-pointing perigee cannot stop the eccentricity growth, but it can control it up to a certain limit. Therefore, the eccentricity will grow and it will stabilize after the first quarter of the year, when the strategy can efficiently control it.

Also from the eccentricity illustrations, the excursions of eccentricity could be seen beyond the circle defined by RE. To better access this problem we can see in Figure 31 that the mean value of the eccentricity is below the control circle.
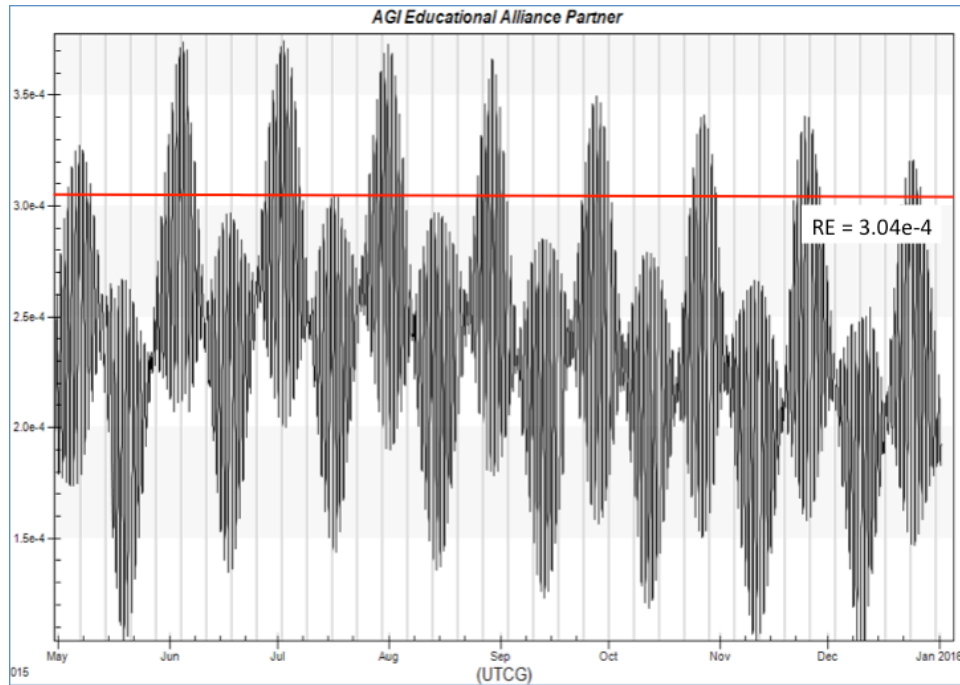
Figure 31.   Eccentricity in time

An excerpt from the eccentricity evolution of the 21x63-day station-keeping cycle is depicted here. Starting in May, we see that the eccentricity crossed the RE value. However, the bulk of the eccentricity drift is below this value. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

In fact, the average eccentricity from May until the end of the simulation was around 2.3e-4.

Because the librations fit inside the proposed control box, no further action was required to manage the eccentricity. However, if the satellite had a greater cross section, the RE would be higher as well, and possibly one could not fit the librations inside the dead band. In this case, a derivation of the sun-pointing perigee called the Solar Lagged Target Strategy can be used; it is explained thoroughly in Hengnian Li [17]. Multiple burns are also an option, but they would penalize the fuel budget.

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2. Inclination Station Keeping

All the satellites were successfully maneuvered in inclination. Regardless of the type of strategy chosen, the fuel expenditure was kept within 1 kg from each other. Table 11 summarizes the north/south station-keeping.

Table 11.   Inclination station-keeping summary

Cycle	Maneuvers	ΔV (m/s)	Fuel (Kg)
74.85º - 7x14 Max interval	26	43.877	34.781
74.85º - 7x14 Zero inclination	26	42.309	33.546
74.85º - 7x21 Max interval	17	42.947	34.049
74.85º - 7x21 Zero inclination	17	41.615	32.998
74.85º - 14x42 Max interval	9	45.01	35.67
74.85º - 14x42 Zero inclination	9	42.189	33.45
74.85º - 21x63 Max Interval	6	41.498	32.909
74.85º - 21x63 Max Inclination	6	41.077	32.576
75º - 21x63 Max Interval	6	41.549	32.92
75º - 21x63 Max inclination	6	41.085	32.585
Mean		42.316	33.548
Standard Deviation		1.221	0.962

The inclination station-keeping varies more than the longitude maneuver. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

Starting with the zero inclination strategy, Figure 32 shows a mosaic containing the station-keeping cycle that did not require cooperation. One obvious thing to notice is that increasing the number of days in the cycle will increase the maximum inclination value. Unfortunately, two maneuvers of the 14x42-day cycle had to target an inclination other than zero, because the satellite would go out of the dead band in the forthcoming cycle.
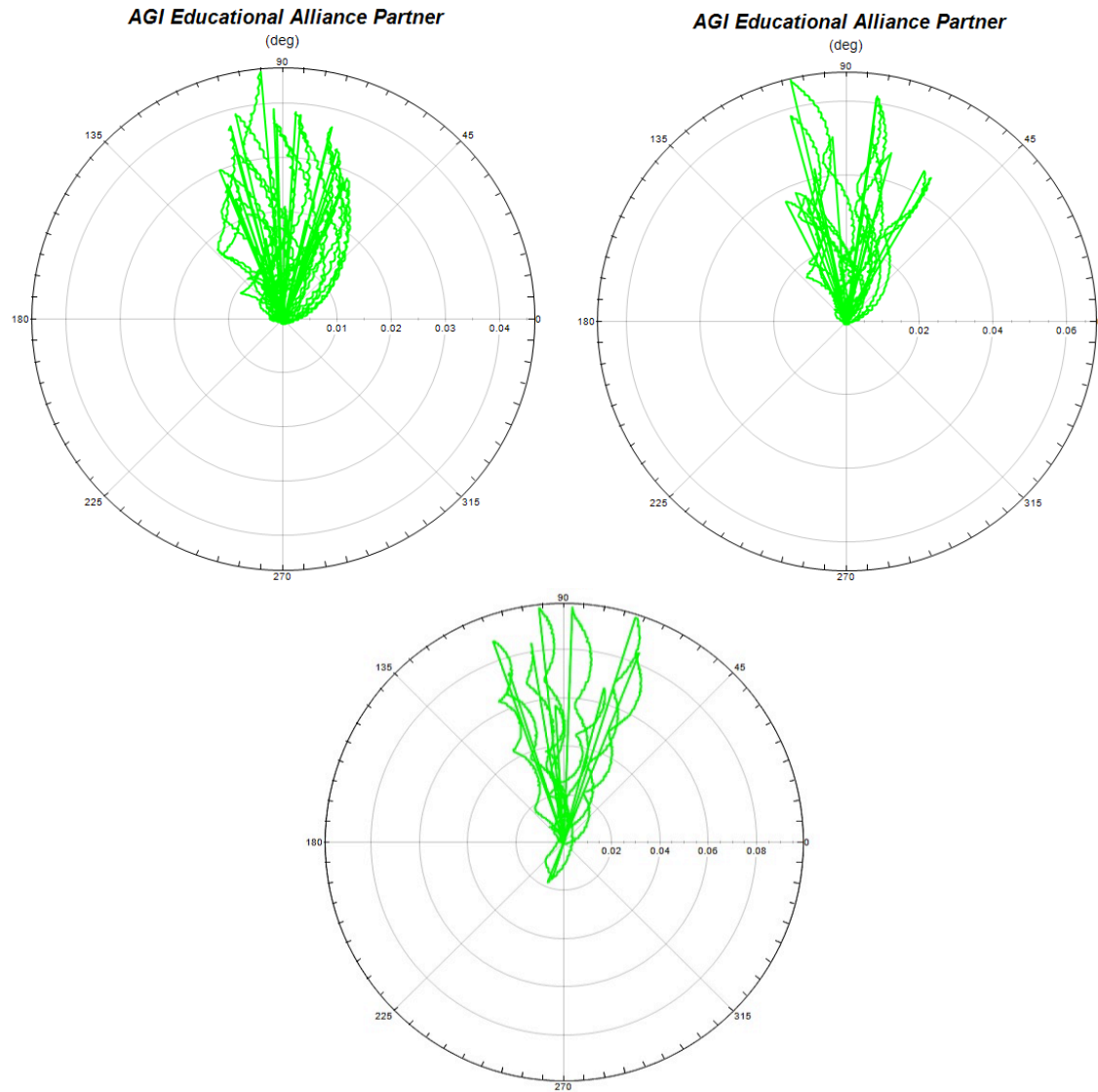
Figure 32.   Zero inclination target

The zero inclination method is shown. From left to right are the 7x14, 7x21, and 14x42-day cycles. In the latter, two maneuvers had to target an inclination of 0.02° to avoid going out of the control box. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

The maximum interval method of the same cycles will follow. The aim here was to make the inclination vector drift through the origin of the circle. Figure 33 presents a mosaic of the inclination station-keeping cycle.
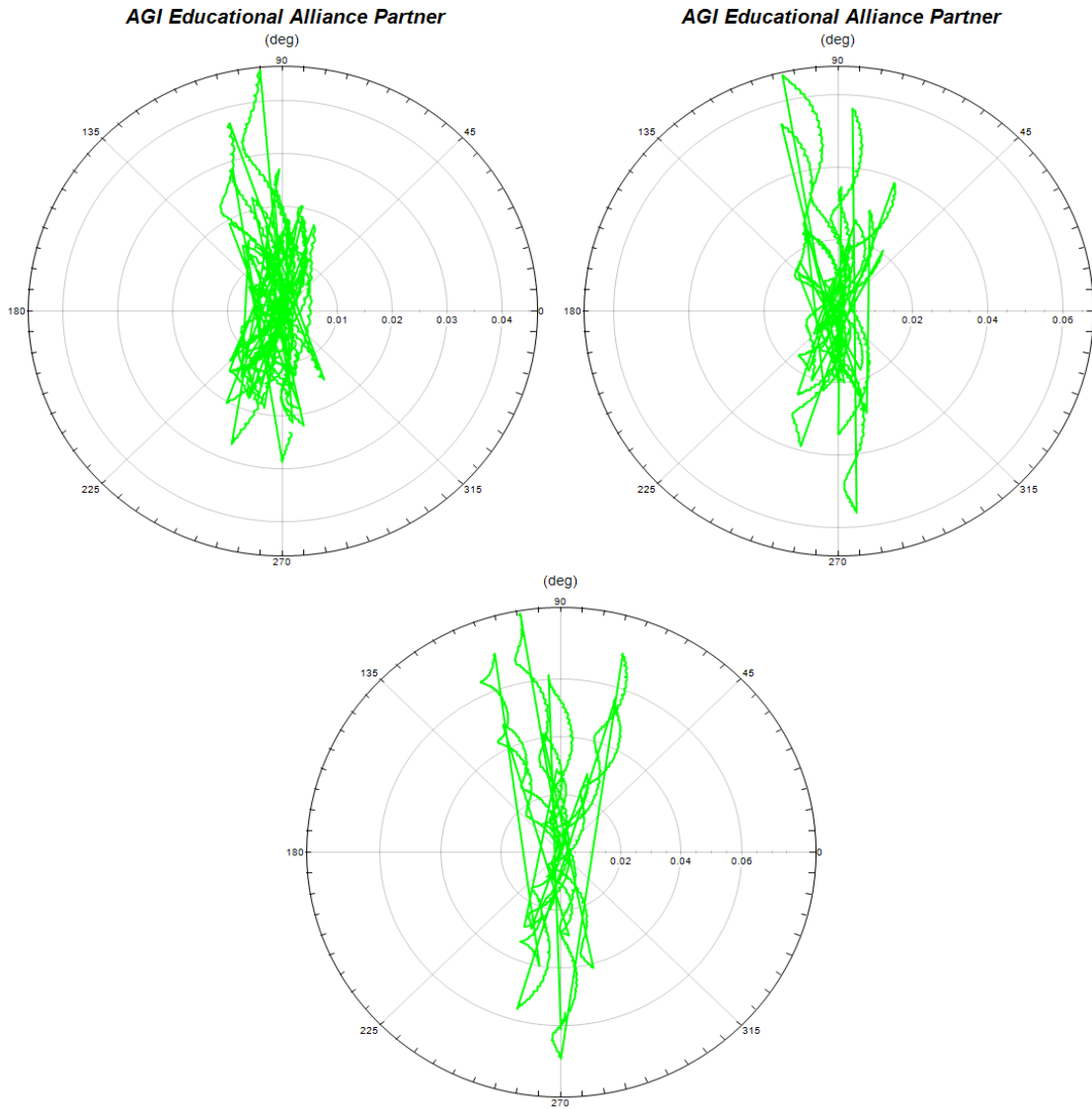
Figure 33.   Maximum interval method

The maximum interval method is shown. From left to right are the 7x14, 7x21, and 14x42-day cycles. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

Lastly, the cooperated satellites will be displayed. Figure 34 presents the maximum interval method and Figure 35 depicts the maximum inclination on the upcoming cycle method.
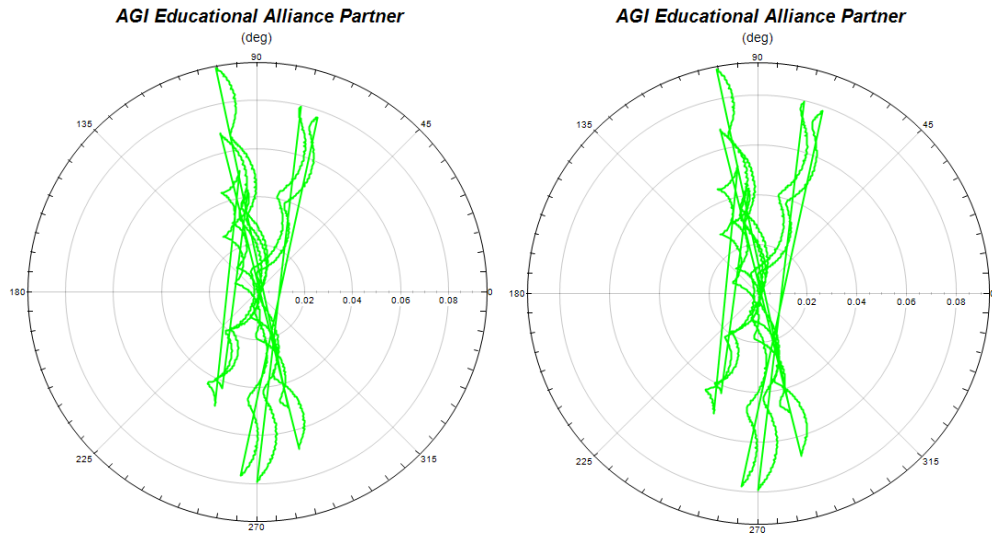
Figure 34.   21x63 cycle maximum interval method

The maximum interval method is shown, for 74.85º and 75º satellites, respectively.
Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.
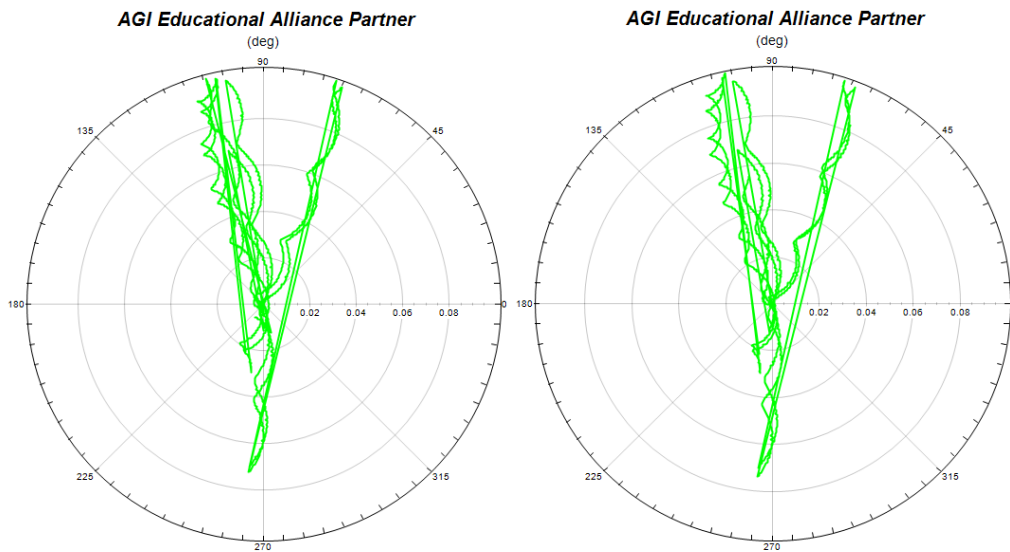

Figure 35.   21x63 cycle maximum inclination method

The maximum inclination method is shown, for 74.85º and 75º satellites, respectively.
Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

3. Overall Results

For overall results, Table 12 shows that the various strategies did not significantly change the fuel expenditure. One aspect to notice is that the maximum interval strategy had larger fuel expenditure than the strategies that had a fixed goal of zero inclination or maximum inclination in the upcoming cycle.

Table 12.   Collocation maneuver summary

Cycle	Maneuvers	ΔV (m/s)	Fuel Mass (Kg)
74.85º - 7x14 Max interval	78	45.131	35.775
74.85º - 7x14 Zero inclination	78	43.562	34.54
74.85º - 7x21 Max interval	69	44.212	35.052
74.85º - 7x21 Zero inclination	69	42.874	33.999
74.85º - 14x42 Max interval	35	46.274	36.673
74.85º - 14x42 Zero inclination	35	43.448	34.451
74.85º - 21x63 Max Interval	24	42.817	33.954
74.85º - 21x63 Max Inclination	24	42.395	33.621
75º - 21x63 Max Interval	24	42.863	33.96
75º - 21x63 Max inclination	24	42.399	33.625
Mean		43.595	34.565
Standard Deviation		1.203	0.948

The overall results are shown. The standard deviation suggests that we have at least 0.94 kg of spread between the maneuvers. Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

IV.   CONCLUSIONS AND RECOMMENDATIONS

This chapter presents some conclusions and recommends further research on the topic. To conclude, we can see that regardless of strategy, we lacked enough data to affirm that one is superior, regarding the fuel optimization. However, one kilo of standard deviation could add up to 15 kg in overall satellite life. In our simulation, the fuel spent with inclination maneuvers made the difference. It is clear that the maximum interval method spent nearly one kilo more compared to other methods. This was a good contrast, showing that we need to follow a clear objective regarding our inclinations targets, for instance, a zero inclination value or a maximum allowed inclination on the forthcoming cycle.

The longitude separation is the simplest collocation strategy that one command center can choose. It has the advantage that satellites are maneuvered independently. As long as they remain inside the dead band, no concerns are expected. In this regard, STK could calculate and predict conjunctions between the satellites, but for our strategy, it would be worthless to use it. Even in the cooperation strategy the close approach was recorded as 60 km. Therefore, the minimum distance between the satellites surpasses the size of the proposed guard band, giving us at least 37 km of separation.

Even though we lack concrete answers regarding the fuel optimization between the strategies, we can enumerate some recommendations.

(1)   Operational Aspects

Every maneuver cycle requires extensive orbit prediction and commitment from the controllers. A small cycle such as the 7x14-day may conflict with payload operations [20], because every week brings another maneuver preparation. The payload can wait, but the satellite cannot, otherwise it would cross the established boundaries and create a collision hazard. As the maneuver cycle is increased, it is obvious that more time will be expended with the payload operation.

From this perspective, the 14x42-day cycle would be the answer if no cooperation were achieved. The 21x63-day is the best option, but cooperation agreement has to account for satellite limitations, command center desires, and communication limitations; sometimes it is hard to reconcile all variables.

(2)   Thrusters utilization

It is known that the Brazilian satellite will have a hydrazine monopropellant propulsion system. Therefore, the thruster will have a common structure with the fuel, hydrazine, being decomposed through a catalyst bed. This catalyst bed is more efficient if it is heated beforehand; therefore, the satellite's controllers could turn the thrusters' electric heaters on before each maneuver. This action would help to avoid the main cause of catalyst bed degradation, the cold-start effect, improving thruster life and efficiency [25, 26]. In thrusters, the rise time will be slightly larger for non-heated catalyst beds than the preheated ones. Regardless, short burns are always less efficient than longer burns, especially the ones that had a cold-start [20]. In other words, longer thrusts would benefit from a more constant thrust profile (Figure 36).
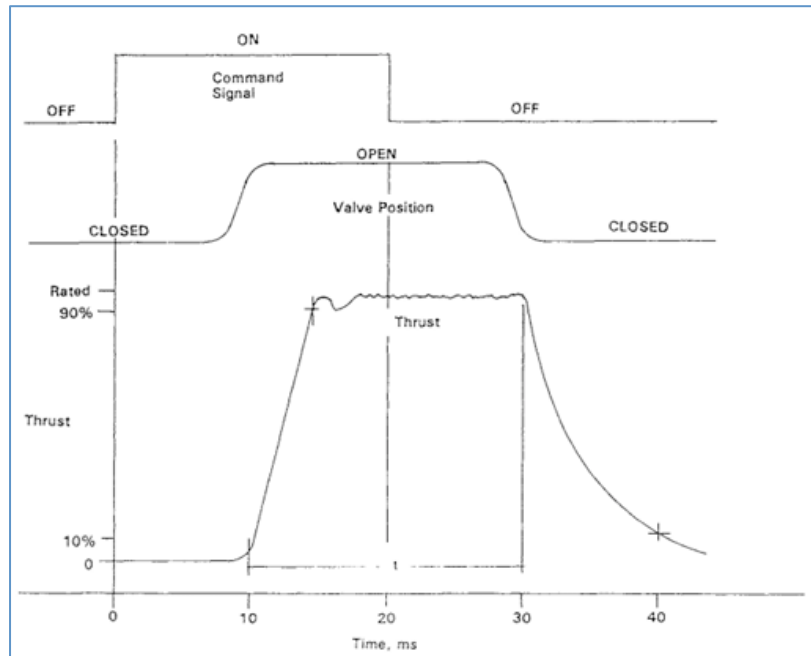
Figure 36.   Thrust profile

A single pulse thrust profile is depicted. The rise time is the time between 10% and 90% of the usable thrust. Longer burns would benefit from longer periods with stable thrust. Source: [25] C. D. Brown, "Monopropellant hydrazine thrusters," in *Spacecraft Propulsion*, 2nd ed., J. S. Przemieniecki, Ed., Washington, DC: American Institute of Aeronautics and Astronautics, 1996, pp. 59.

Another topic of thruster utilization would be system reliability. Reliability is defined as the probability that a component will maintain its functionality through a specific period or the extent of operation [27]. All thrusters have a reliability value that could impact the design life of a satellite. Using the system less frequently can increase its operational life, and make it less susceptible to failure. Therefore, reducing thrusters utilization can increase the overall mission reliability. Figure 37 demonstrates some common reliability concepts:
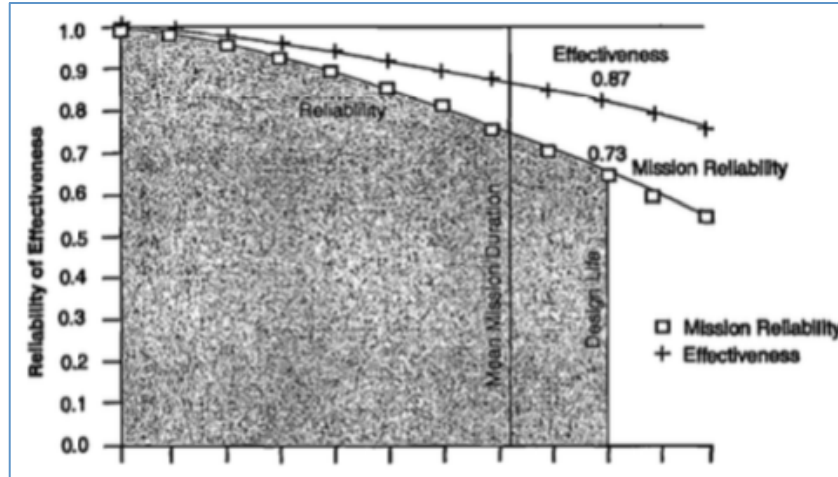
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Figure 37.   Mission reliability curve

Increasing the reliability of a component drags the reliability line up, thus increasing the mission reliability at the expected design life. Source: [28] J. R. Wertz and W. J. Larson, *Space Mission Analysis and Design*, 3rd ed. Dordrecht, Netherlands: Kluwer Academic Publishers; Torrance, CA: Microcosm Inc., 1999, pp. 767.

With this assessment, the 21x63-day cycle would be the best choice, but again this would require a cooperative agreement between centers. The 14x42 appears to be the best option for a stand-alone operation.

A.   FURTHER WORK

The objective of this study was to research an optimization for the complete separation longitude strategy. However, further research in new strategies for the same problem would be beneficial. Following are my suggestions.

One can assume that a control center has under its command two satellites and needs to collocate them in the same dead band proposed in the collocation agreement. This scenario could be favorable for an inclination and eccentricity strategy. However, another complete longitude separation, with tight station-keeping cycles, could also be modeled.

The simulation itself could be improved, seeking a possible automation for the Astrogator commands. As stated, the longitude targets at the edges of the dead band were

manually found; an effort to command the software to perform this duty would be a better approach to increase reliability of the study.

Another area of improvement would be the fidelity in which a satellite can be modeled. In our simulation we have used generic performance data. Specific engine performance, thrusters position, reflection coefficient, spacecraft mass and cross-section area would increase the confidence in the simulation as well.

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APPENDIX. STK TUTORIAL

This tutorial was devised to allow the reader to follow and recreate the simulation. It is a set of steps to configure a scenario, and assess the station-keeping maneuvers. This material was adapted from the "GEO Maintenance" tutorial on the Astroguild website [5].

A.   SCENARIO CONFIGURATION

First, configure and populate the scenario.

Configure the scenario properties:

1. On the object browser, double click on the scenario icon.
    a. Under Time, configure the "Start" to 1 Jan 2015 00:00:00, and "Stop" to 10 Jan 2016 00:00:00.
    b. Using the Units, change the Distance unit to "Meters (m)" and Time to "Days."
    c. Click on OK.

Create the dead band:

1. Create a new satellite named "74.85_Box":
    a. Using the orbit wizard, create a geosynchronous satellite centered at the -74.85 GEO slot.
    b. Leave the ephemeris spanning the scenario analysis.
    c. Open the properties of the satellite and set the "Coord. System" to be "TrueOfDate" and then set the "Time Step" to 20 Min.
    d. Apply Changes.
2. Under Basic, Orbit, use the "TwoBody" propagator.
3. Using the 2D Graphics, Attributes, and "Basic" setting, de-select "Show."
4. Using the 3D Graphics, Proximity, "Geostationary Box" setting:
    a. Create a Geostationary box at -74.85º, a NS of 0.1º, and an EW of 0.05º.
    b. Select a desired color for the dead band.

5. In an Earth Fixed window, focus on the control box by centering on the -74.85°
    Satellite.

    a. Use the "View From/To" button, and select the -74.85°W satellite to "View
        From" and the Earth to "View to."

    b. In the 3D Graphics/Model page, turn off the model and de-select "Show"
        under Marker.

Note: You may want to store this view *. Do this for every satellite you want to create. It will expedite further
visualizations.*

Create some "Calculations Objects." Click on the UTILITIES menu, and then on
the COMPONENT BROWSER:

1. Create an object called Inclination TOD.

    a. Click on "Calculation Objects" and then on "Keplerian Elems."

    b. Click on "Inclination," duplicate it, and name it "Inclination TOD."

    c. Double click "Inclination TOD":

    d. Change the "CoordSystem" to Earth TOD.

    e. Click Ok.

2. Create an object called RAAN TOD.

    a. Click on "RAAN," duplicate it, and name it "RAAN TOD."

    b. Double click "RAAN TOD":

    c. Change the "CoordSystem" to Earth TOD.

    d. Click Ok.

3. Create an object called TOD Inc Diff.

    a. Still under "Calculation Objects," click on "Math."

    b. Click on "Difference," duplicate it, and name it "TOD Inc Diff."

    c. Double click "TOD Inc Diff":

    d. Change the "CalcObject" to Inclination TOD, under "Keplerian Elems."

    e. Click Ok.

4. Create an object called RAAN Diff.

    a. Click on "Difference" again, duplicate it, and name it "RAAN Diff."

    b. Double click "RAAN Diff":

c. Change the "CalcObject" to RAAN TOD, under "Keplerian Elems."

d. Click Ok.

5. Create an object called Longitude Constraint.

    a. Click on "Constraint" at main root of the browser.

    b. Click on "UserDefined," duplicate it, and name it "Longitude."

    c. Double click "Longitude":

    d. Change the "CalcObject" to Longitude under the "Geodetic" element.

    e. Set "Value" to -74.9º and "Tolerance" to 0.01º.

In this tutorial, we will first build a sequence that keeps us within an East-West "box." Then, we will build a similar North-South sequence, which maintains our satellite within the Northern and Southern boundaries of the box. We will define our "box" to be 0.1º north and south of the equator, and 0.05º east and west of the 74.85º W reference point.

Create a satellite to control:

1. Insert a default satellite.

2. Change its name to "74.85_7X14_Days."

3. Using the Propagator, select Astrogator.

    a. Under Elements:

        i. Change the Coord. System to TOD ("TrueOfDate").

        ii. Select the coordinate time to "Keplerian."

        iii. Change the "Semi-major Axis" to 4.21665e+007m.

        iv. Set the "Eccentricity" and "Inclination" to 0.

        v. Change the "Right Asc. of Asc. Node" to "Lon. of Asc. Node" and set it to 285.345º.

    b. Under Spacecraft Parameters:

        i. Change the "Dry Mass" to 2200 Kg.

        ii. Change all the areas to 50m^2.

        iii. Change the "Solar Radiation Pressure" Coefficient to 1.3.

    c. Under Fuel Tank:

        i. Change Fuel Mass to 150 kg.

d. Using 3D Graphics:

    i. Under Pass, "Orbit Track" settings, change the trail type to percent, and select 100.

    ii. Under Orbit System, check the box of the "Fixed by Window."

    iii. Under Vector, select the "Sun Vector."

    iv. Under "Options," check "Draw at a Point" and select the satellite center.

    v. Under Common Options, change the "Scale" to 4.0000.

    vi. Click on "Add" to add a Vector Component:

    vii. From the available list, select the satellite, look for the "Ecc" icon and move it to the "Select" box. Click on OK.

    viii. Select "Ecc Vector" and check the box "Draw at Central Body."

e. In the Model page, turn off the model and de-select "Show" under Marker tab.

f. In Data Display, select to show "Classical Orbit Elements" and add the "Fixed LLR Position" from the "Add" button.

    i. Click on "Fixed LLR Position." Set its "Position," the "Y Origin" to bottom.

    ii. Apply to both:

        1. Under Appearance, change "Format" to "No Labels" and de-select "Title."

g. Click Ok.

Tip: To simulate more than one satellite, copy and paste the above satellite and change its name. This action will save some time.

B.   EAST-WEST STATION KEEPING

Since we know that our satellite will drift to the west, we will first create a "Sequence" MCS segment that contains a targeter that causes the satellite's drift to reverse direction (i.e., go from an eastern to a western drift) at the halfway point to our desired station-keeping days. But first propagate the satellite to the western edge:

1. Double click on the propagate segment:
   a. Change its name to Propagate to the Edge.
   b. Change the Coord. System to Earth TOD.
   c. Change the color to light blue (Optional).
   d. Click Ok.
2. Under Stopping Conditions, click on "New."
   a. Double click on "AscendingNode."
   b. Change the "Repeat Count" to 5 and the "Cord. System" to TOD.
   c. Click on "Duration" and delete it.
3. Click on "Add to Component Browser" ![icon]. This action will save this segment into the Component Browser and accelerate some future configurations.

Note: The Propagate to the Edge segment will propagate the satellite forward, closer to the edge. Therefore, Astrogator will repeat the stopping conditions (Ascending Node) five times and then stop the propagation.

4. Insert a Sequence segment after the last Propagate segment, and name it "SK 14 days." Inside "SK 14 days" insert another Sequence and name it "EW SK 7 Days."
5. Inside the last sequence, add a propagate segment, double-click on it and change its color to red, and name it "Propagate to Sun Point." Click Ok.
   a. Add a stopping condition to UserSelect. Change its name to Sun Pointing Longitude.
   b. Under User Calc Object, click on "Edit," "Change" and choose "Local Apparent Solar Longitude" under the "Other Orbit" folder.
   c. Set "Trip" to 90.
   d. Set "Tolerance" to 0.1º
   e. Under Constraints click on ![icon], select "Longitude," move it to the right, and click OK.

This segment will propagate to a place 90º relative to the sun that occurs before our satellite's longitude has drifted out of our "box" on the western side.

6. Add a Target Sequence. Double click it, name it "Turn Around" and change "Coord. System" to "Earth TOD."

    a. Change the "Action" to "Run active profiles."

    b. Insert a Maneuver element, name it "Drift Maneuver," change its color to green, and the "Coord. System" to "Earth TOD."

        i. Leave "Maneuver Type" as Impulsive.

        ii. Change the "Attitude Control" to Thrust Vector.

        iii. Set the "X (Velocity)" to -0.04 m/s (remember to change the units), and check the target button .

        iv. Under the "Engine" tab:

            1. Leave the "Engine Model" as Constant Thrust and Isp.

            2. Check the "Update Mass Based on Fuel Usage" box.

Note: This is the place to change propulsion type and insert engine specifications, if desired.

    c. Insert a Propagate element, change its name to "Propagate 7," the color to light blue, and the "Coord. System" to "Earth TOD."

        i. Change the "Trip" to 6.8 days, and check the target button .

        ii. Leave the "Tolerance" as it is.

   d. With "Propagate 7" highlighted, click on the results button:
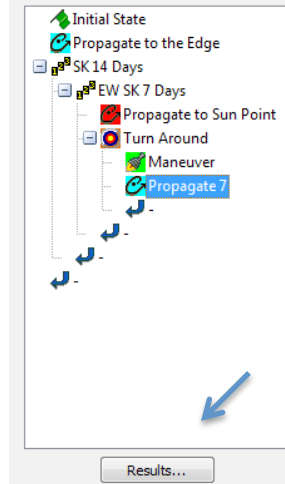
Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

      i. Under "Geodetic," select "Longitude," and move it to the right.

      ii. Under "GeoStationary," select "Longitude Drift Rate," and move it to the right. Click Ok.

7. Go back to the target "Turn Around":

    a. Change the name "Differential Corrector" to "EW SK."

    b. Click on Properties 📋.

    c. Under "Control Parameters," check all the boxes.

        i. Under the Impulsive maneuver:

            1. Check the box "Custom Display Unit" and select m/sec under the "Display Unit" drop menu.

            2. Set the "Perturbation" to 0.001 m/sec.

            3. Set "Max Step" to 0.1 m/sec.

            4. Leave the "Trip Value" as it is.

        ii. Under "Equality Constraints," check all the boxes.

            1. Set the "Longitude" to -74.89º.

               a. Set the "Tolerance" to 0.0001º.

               b. Set the "Longitude Drift Rate" to 0.0016643 deg/day.

    d. Click Ok.

Note: The longitude value on the "Equality Constraints" will have to be changed manually every station-keeping in order to find the closest point to the west edge.

Run the MCS. ⬛. The propagation of the satellite and the target working to find the established goals should be visible. The end product should look like this:
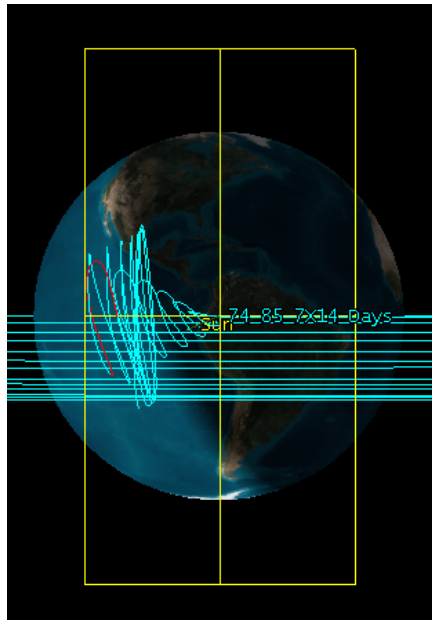
Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

After the target has been converged, change the MCS to "Run Only Changed Segments" by clicking on the drop arrow:
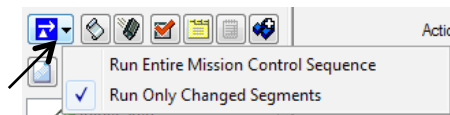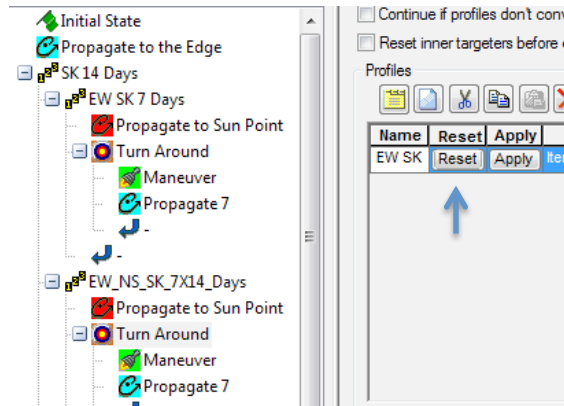

Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

C. NORTH-SOUTH STATION KEEPING

We will create a "nested" target sequence. The Inclination station-keeping will occur close to end of a longitude station-keeping.

1. Copy "EW SK 7 Days," and paste it after the first copy, inside the "SK 14 Days" sequence.

2. Change the name of the copy to "EW NS SK 7x14 Days." Reset the Profile under the "Turn Around" targeter of the new copy.

Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

3. Insert a Propagate segment before "Propagate 7" segment. Change its name to "Propagate 3," change the color to green, and the "Coord. System" to TOD.
   a. Set the "Trip" to 3 days, the "Tolerance" to 1e-10 days and check the target button.

4. Between the "propagate" segments:
   a. Insert a new Target Sequence. Double click it, name it Inclination and change the "Coord. System" to "Earth TOD."
      i. Change the "Action" to Run active profiles.
      ii. Insert a Propagate element. Change its name to "Propagate to Ignition," change the color to purple and the "Coord. System" to TOD.
         1. Set the "Trip" to 1 day, and check the target button.
      iii. Insert a Maneuver element, name it Maneuver, and the "Coord. System" to Earth TOD.
         1. Leave "Maneuver Type" as Impulsive.
   2. Change the "Attitude Control" to Thrust Vector.

3. Set the "Y (Normal)" to 6 m/s (remember to change the units), and check the target button .

iv. Under the "Engine" tab:

1. Leave the "Engine Model" as Constant Thrust and Isp.

2. Check the "Update Mass Based on Fuel Usage" box.

v. With "Maneuver" highlighted, click on the results button:

1. Under "Math," select "RAAN Diff," "TOD Inc Diff," and move them to the right.

2. Under "Keplerian Elems," select "Inclination TOD," "RAAN TOD," and move them to right.

3. Click Ok.

b. Go to the target "Inclination":

i. Change the name "Differential Corrector" to "Target RAAN."

ii. Click on Properties .

1. Under "Control Parameters," check all the boxes.

a. Under the Impulsive maneuver:

i. Check the box "Custom Display Unit" and select m/sec under the "Display Unit" drop menu.

ii. Set the "Perturbation" to 0.1 m/sec.

iii. Set "Max Step" to 10 m/sec.

b. Leave the Trip Value as it is.

2. Under "Equality Constraints," check all the boxes that apply.

Here we set the values, so the user can choose an inclination station-keeping.

a. Set the "Inclination TOD" to the desire value of inclination. Example: 0.001°.

i. Set the "Tolerance" to 0.01°.

b. Set the "TOD Inc Diff" to the desired value of inclination. Example: 0º to have the same value as before the maneuver.

    i. Set the "Tolerance" to 0.01º.

c. Set the "RAAN TOD" to the desired value of RAAN. Example: 270º.

    i. Set the "Tolerance" to 0.01º.

d. Set the "RAAN Diff" to the desired value of RAAN. Example: 180º to invert the RAAN 180º.

    i. Set the "Tolerance" to 0.1º.

3. For demonstration purposes, check "Inclination TOD" and "RAAN TOD" with the above-mentioned values.

4. Click Ok.

c. Click on "Propagate 7."

    i. Change its name to "Propagate 3_2" and its color to yellow.

    ii. Set the "Trip" to 2.8 days.

    iii. Set the "Tolerance" to 1e-005 day.

d. Go back to the "Turn Around" targeter:

    i. Click on Properties .

    ii. Under "Control Parameters," check the "Propagate 3_2" box.

    iii. Click on "Hyde Inactive" on both panels.

5. Run the MCS.

The targets working together to get a solution should be visible. Eventually, the following will appear:
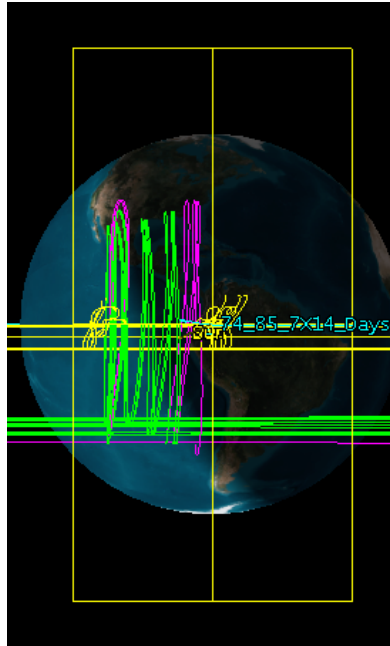
Source: [12] *Systems Tool Kit*, version 10.1.3. Analytical Graphics Inc., Exton, PA, 2015.

Note: The inclination target may not converge in some station-keeping cycles. Try to change the direction of the normal velocity ("Maneuver" element under the "Inclination" target). With a negative sign, STK will seek the descending node.

Take a better look at the satellite clearing the graphics of Astrogator by clicking on . Play the simulation to see the satellite in action.

Save the sequence:
1. Reset all targets profiles.
2. Click on "SK 14 Days" and add it to the component browser .

With the existing sequence one can derive any other station-keeping sequence. To continue with the simulation:
1. Run the MCS.
2. Insert a new "SK 14 Days" sequence from the component browser after the last sequence as necessary.

Now you can create any type of station-keeping cycle. Keep on doing these procedures and do not forget to save.

The file that has been used for this thesis simulation is available on Google Drive:

https://drive.google.com/folderview?id=0BwkRNkq1Dx1uVUNhbHJDVzRueGs&usp=sharing

User: GeoSKSimul@gmail.com

Password: NPSSpaceOps

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